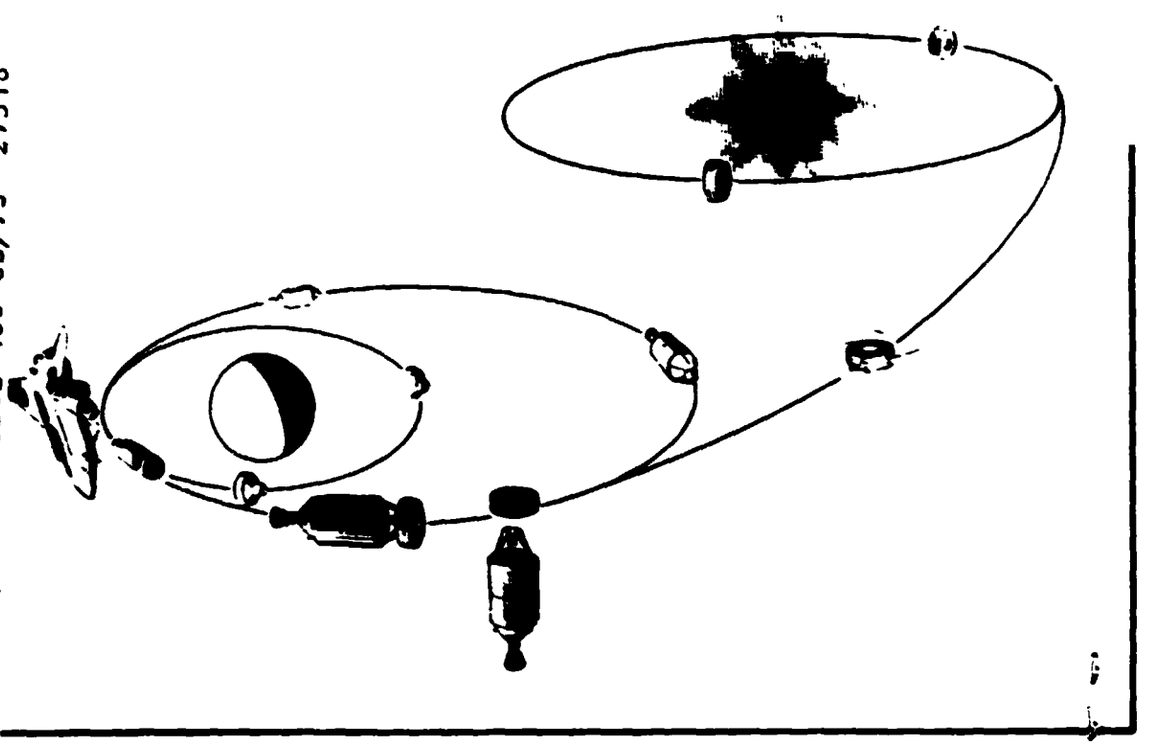
ANALYSIS OF SPACE SYSTEMS STUDY FOR THE SPACE DISPOSAL OF NUCLEAR WASTE

(NASA-CR-161867) ANALYSIS OF SPACE SYSTEMS STUDY FOR THE SPACE DISPOSAL OF NUCLEAR WASTE. STUDY REPORT, VOLUME 1: EXECUTIVE SUMMARY Final Report (Boeing Aerospace Co., Seattle, Wash.) 65 p HC A04/MF A01 CSCL 18G G3/13 27518 Unclas



STUDY REPORT - VOLUME I

EXECUTIVE SUMMARY

1981

D180-26426-1

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Contract NAS8-33847
DPD 590
DR 4

THE BOEING COMPANY

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Submitted to

The National Aeronautics and Space Administration

George C. Marshall Space Flight Center

By

Boeing Aerospace Company
Seattle, Washington 98124

FOREWORD

The study results described in this report are a part of an ongoing analysis to determine the feasibility and preferred approaches for disposal of selected high-level nuclear wastes in space. The Boeing Aerospace Company (BAC) study is an integral part of the ongoing Office of Nuclear Waste Isolation (ONWI) managed DOE/NASA program for study of nuclear waste disposal in space. The research effort reported here was performed by the Boeing Aerospace Company Upper Stages and Launch Vehicles Organization under NASA Contract NAS8-33847 from May of 1980 until March of 1981. The study objective was to identify, define and evaluate reasonable alternative concepts for the space disposal of nuclear waste, selecting alternative concepts of high merit for further evaluation, and documenting the evaluation and selection process.

The information developed during the study period is contained in this two-volume final report. The title of each volume is listed below:

Volume I Executive Summary
Volume II Technical Report

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ACKNOWLEDGMENTS

The author wishes to acknowledge the generous assistance of the following individuals who provided significant contributions to the technical content of this report:

J. C. Jenkins:	o mission analysis o orbit transfer system concept development o cost and programmatics
D. Grim:	o electric propulsion
H. Whippo:	o propulsion system analysis
R. T. Conrad:	o structures and weights
C. L. Wilkinson:	o thermal analysis o aerothermodynamic analysis
H. Zeck:	o aerodynamics
R. J. Gewin:	o avionics o electrical power
A. R. Hathaway (BEC)*:	o BEC study manager o radiation shield trades
J. L. Anderson (BEC)*:	o waste payload protection options and trade studies o waste payload configurations
M. P. Baze:	o radiation shield analysis
I. Arimura:	o radiation effects on avionics

* Boeing Engineering & Construction Co.

Acknowledgment is also made to Mr. Claude C. (Pete) Priest, the NASA/MSFC contracting officer representative for the bulk of the study, for the outstanding guidance and cooperation provided; to his replacement, Mr. William Galloway, for valuable support during preparation of the final reports and to various members of the MSFC staff, particularly Robert F. Nixon and Rowland E. Burns. Also, special thanks must go to Ralph E. Best, Manager of Alternative Concepts at the Department of Energy (DOE) Office of Nuclear Waste Isolation (ONWI), Columbus, Ohio, and Philip R. Compton, Energy Systems Division, Office of Aeronautics and Space Technology, NASA/Headquarters, Washington, D.C., for their guidance throughout the study.

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1.0 INTRODUCTION

Since 1970 a number of concepts for space systems for nuclear waste disposal have been studied and evaluated. This study has integrated the results of these previous studies in a systematic fashion to identify and document viable alternative space system concepts having high merit. This effort is an integral part of the ongoing NASA/DOE program for evaluation of the space option for disposal of certain high level nuclear wastes in space as a complement to mined geologic respositories. This introduction provides a brief overview of the study background, scope, objective, and approach.

1.1 BACKGROUND. The need to isolate safely or dispose of nuclear waste materials is a problem for this nation and other nations of the world. This problem has been studied for many years. Comprehersive NASA studies of space as a disposal site did not begin until early 1970's when the NASA Lewis Research Center (LeRC) began studies which included comprehensive design analyses and concept testing of a nuclear waste payload that could survive Earth atmospheric reentry and impact from space.

In 1975, the NASA's Marshall Space Flight Center (MSFC) initiated a separate series of studies of the disposal in space of nuclear waste. These studies, and studies conducted by the NASA Ames Research Center in 1976, were followed by the 1979-80 study program conducted by MSFC and the Battelle-Columbus Laboratories.

These efforts provided the basis for the preparation of the comprehensive system safety design requirements summarized in Appendix E of Volume 2. These requirements, in combination with the data on space disposal destinations provided by the earlier studies, have provided for the first time a systematic and comprehensive set of requirements for evaluation and design of space systems for the disposal of nuclear waste. These requirements, together with the extensive data base on space systems provided by the MSFC 79-80 study efforts have provided the basis for this study.

1.2 OBJECTIVES. The overall objectives of this study were identification and definition of space systems concepts, evaluation of these concepts as to their

performance, risk, and technical viability, and selection of the most attractive space system concepts for further consideration.

To accomplish these overall study objectives, the study was divided into eight major study areas, each having its own objectives. These objectives are defined below for each study task:

Task 1: Mission and Operations Analysis (Section 5.2)
- Define orbit transfer system trajectories and performance requirements.
- Define mission operations and functional requirements.
- Identify mission control requirements.

Task 2: Waste Payload Systems (Section 5.5)
- Identify protection system requirements.
- Trade full range of protection system options for containment, radiation shielding, re-entry protection, and impact protection.
- Select and define waste payload and protection systems.

Task 3: Flight Support System (Section 5.6)
- Identify flight support system requirements.
- Define and evaluate flight support system concepts.
- Characterize selected concepts.

Task 4: Launch Site Systems (Section 5.8)
- Define impact of selected space disposal operations on KSC.
- Select and define alternate launch site.
- Compare and evaluate remote site option.

Task 5: Launch Vehicle Systems (Section 5.4)
- Identify viable launch system options.
- Trade launch options to select optimum launch system for space disposal.
- Define candidate systems.
- Identify unique requirements.

Task 6: Orbit Transfer System (Section 5.3)

- Identify orbit transfer system options (propulsion, staging, reuse).
- Trade options to determine optimum candidate systems.
- Characterize selected systems.
- Identify rescue mission requirements.
- Define rescue systems.

Task 7: Space Disposal Destinations (Section 5.1)

- Identify and characterize candidate destinations.
- Trade candidates against the reference 0.85 AU heliocentric orbit destination to determine if safer or cheaper alternatives exist.

Task 8: Systems Integration and Evaluation (Section 5.7)

- Integrate systems from Tasks 2, 3, 5, and 6 into total system concepts.
- Define candidate systems in terms of cost and risk.
- Trade total system concepts in terms of risk and cost to select alternative systems of high merit for further study.

1.3 SCOPE. This study covers the systematic identification, definition and evaluation of reasonable space system concepts leading to the integration and evaluation of total system concepts for space disposal of nuclear waste. Specific study areas included space destinations, space transportation options, launch site options, nuclear waste payload protection approaches, and payload rescue techniques. Maximum use was made of the previous studies and assessments of the space disposal of nuclear waste. Definition of the space system concepts was on a common basis, consistent with the study ground rules. Additional analyses and definition were performed on space system concepts not thoroughly assessed in previous studies. Total system concepts resulting were evaluated for performance and risk, leading to selection of four concepts of high merit for further evaluation.

1.4 APPROACH/GUIDELINES. The overall approach used in conducting the study is illustrated schematically with a summary of key inputs and output in Figure 1.4-1. Tasks are shown in the order they were accomplished. In Task 8, the

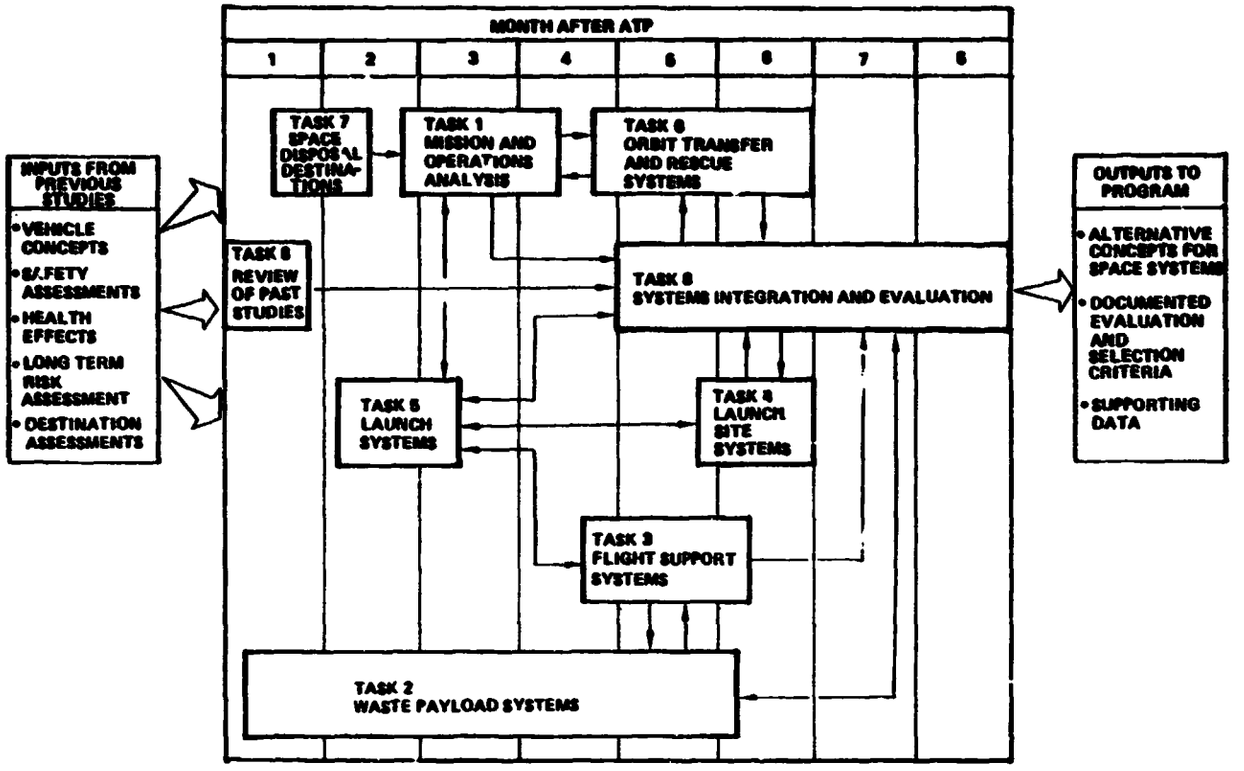

Figure 1.4-1: Overall Approach Showing Task Interrelationships

initial effort reviewed past studies to identify and assess past space system concepts. A preferred destination for the study was established in Task 7, allowing definition of mission characteristics and operations in Task 1. Launch systems were characterized and traded in a parallel effort in Task 5. In Task 6, orbit transfer systems were defined and traded to select the best systems for the mission characteristics defined in Task 1. A parallel effort in Task 2 defined the characteristics of waste payload and waste payload protection systems; these results, with the launch vehicle characterization provided by Task 5 allowed definition of flight support systems in Task 3. Waste payload characteristics from Task 2, Flight Support Systems from Task 3, Launch Systems from Task 4 and Orbit Transfer Systems from Task 6 were integrated into total system concepts and traded to select the alternative systems of highest merit in Task 8. A final effort in Task 3 assessed the impact of the selected options on the Kennedy Spacecraft Center and an alternate Launch Site.

Significant guidelines and assumptions that were used in the study are as follow:

1. Maximum use was made of past studies and other associated data as appropriate.

2. The reference concept for initial nuclear waste disposal in space from the 1979-80 study activity described in Section 1.5 was used as a starting point for definition of nuclear waste characteristics (mix, form, quantity).

3. Cost estimates for elements of the space systems were expressed in 1980 dollars.

4. Containment and system safety requirements used as a starting point for the waste payload systems task were as defined in the 1979-80 Reference Concept. Requirements were reviewed and modifications recommended as appropriate.

5. Operational waste disposal flights were assumed to start in the 1990-1995 time period.

6. Where appropriate, projections of normal growth technology to 1990 levels were used in design of vehicles.

7. Technical work guidelines, considerations, and assumptions as specified in the Study Plan were followed.

1.5 **REFERENCE NUCLEAR WASTE DISPOSAL SYSTEM DESCRIPTION.** Because of its key role as a point of departure for definition of candidate concepts in all study areas and as a standard for performance and risk evaluation of alternate total systems for space disposal, a brief description of the MSFC reference concept for nuclear waste disposal in space is included in this section. A comprehensive description of the reference concept is presented in Appendix D of Volume 2, The Technical Report. The major aspects of the reference mission are illustrated in Figure 1.5-1. This mission profile has been divided into seven major activities. The first two are expected to be the responsibility of the Depart-

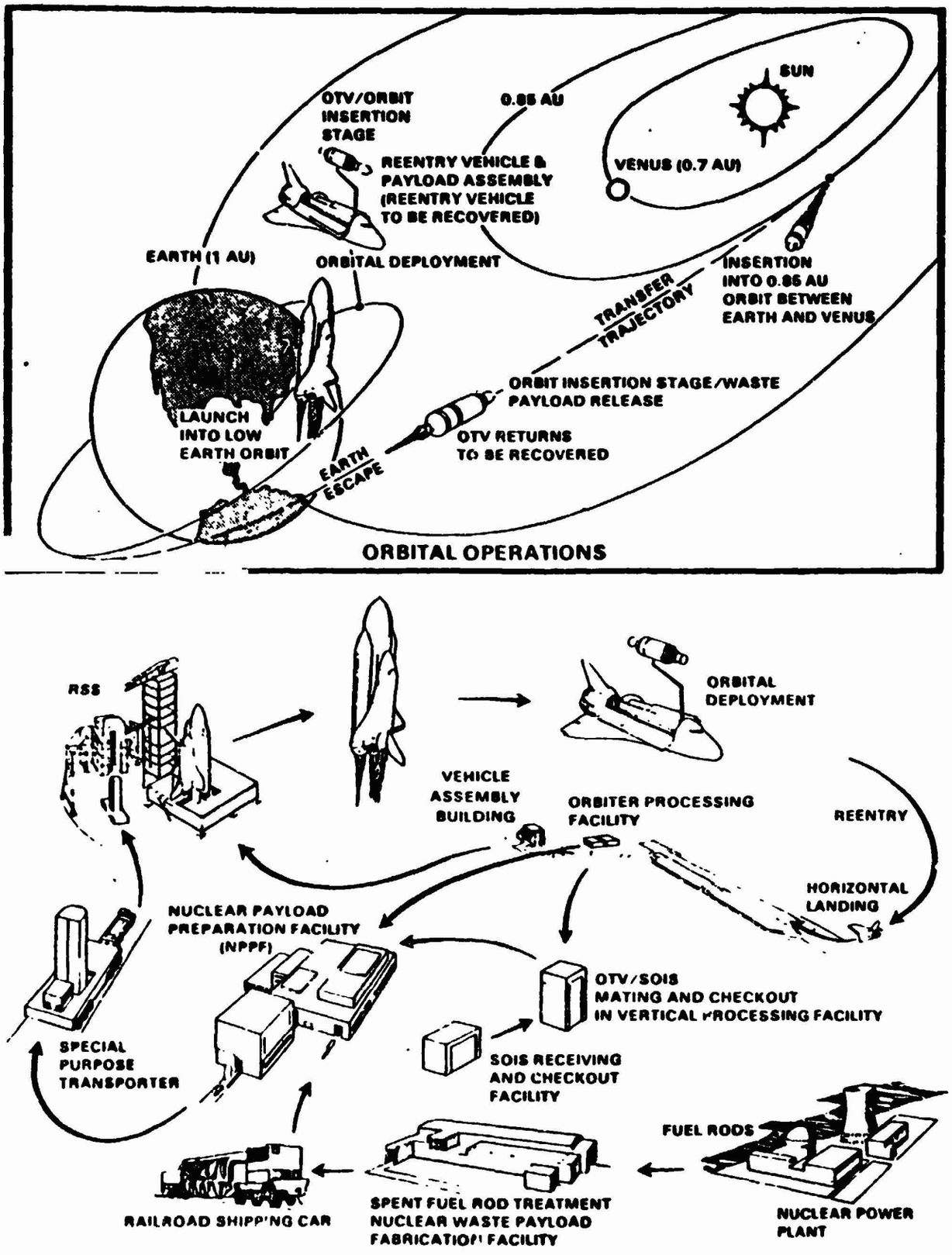

Figure 1.5-1: Ground and Space Operations for Reference Space Disposal Mission

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ment of Energy (DOE) and the last five are expected to be NASA's. These are:

1) Nuclear Waste Processing and Payload Fabrication (DOE)
2) Nuclear Waste Ground Transport (DOE)
3) Payload Preparation at Launch Site (NASA)
4) Prelaunch Activities (NASA)
5) Uprated Space Shuttle Operations (NASA)
6) Upper Stage Operations (NASA)
7) Payload Monitoring (NASA)

Only activities 5, 6 and 7 were considered in detail in this study; the description presented here emphasizes these activities. Descriptions of the remaining aspects can be found in Appendix D of Volume 2.

Prelaunch Activities. Following nuclear waste processing and payload fabrication, the waste container is transported by rail to the nuclear payload preparation facility (NPPF) located at the launch site. The NPPF is expected to provide interim storage capability for up to three shielded waste containers, which affords efficient preparation for launches plus capacity for unplanned delays. During storage, additional radiation shielding, thermal control, monitoring and inspection of the waste container would be provided. The payload is transferred from the NPPF to the pad after the Shuttle vehicle installation at the launch pad has been completed. The payload is then positioned by the RSS and installed in the Orbiter cargo bay. After payload installation, propellant loading of the OTV, and final systems checkout, the decision to launch is made.

Uprated Space Shuttle Operations. One Uprated Space Shuttle vehicle (LOX/RP-1 reusable boosters replacing the solid rocket boosters) would be readied for launch for a given disposal mission. The Uprated Space Shuttle (45,400 kg payload to low Earth orbit) that is to perform the disposal mission is launched from KSC at a 108 degree south azimuth to a 300 km (160 nmi) circular orbit inclined 38 degrees to the equator. Once on orbit, the loaded reentry vehicle (RV) in the Shuttle Orbiter cargo bay is remotely translated aft a short distance and structurally latched to the SOIS. Using the OTV payload bay rotation structure, the OTV, SOIS, and loaded RV are deployed from the Orbiter bay. After the configuration has been stabilized in a fixed attitude, the Orbiter

will move to a safe distance away to limit the radiation dose to the crew from the unshielded payload. At this time, the waste payload would be mechanically transferred by remote control to the SOIS payload adapter, and the OTV/SOIS/waste payload is oriented for the Earth escape propulsive burn. The reentry vehicle would remain in orbit and be recovered and returned to KSC by the Shuttle Orbiter.

Upper Stage Operations (NASA). After the OTV/SOIS/waste payload system has passed final systems checkouts, the OTV propulsive burn would place the SOIS and its attached waste payload on the proper Earth escape trajectory. Control of the propulsive burn from low Earth orbit would be from the aft deck payload control station on the Orbiter, with backup provided by a ground control station. After the burn is complete, the SOIS/waste payload is then released. In about 160 days the payload and the storable liquid propellant SOIS would travel to its perihelion at 0.85 A U about the Sun. (One astronomical unit is equal to the average distance from the Earth to the Sun.) The SOIS would then place the payload in its final space disposal destination by reducing the aphelion from 1.0 to 0.85 A U. To aid in obtaining the desired orbital lifetimes, this orbit would be inclined to the ecliptic plane by 1 degree. The recovery burns of the OTV would use the remaining OTV propellant to rendezvous with the Shuttle Orbiter for its subsequent recovery, refurbishment, and reuse on a later mission (see Figure 1.5-1).

Payload Monitoring. The Earth escape trajectory of the SOIS/waste payload would be monitored by ground-based radar systems and telemetry from the SOIS and OTV. The final disposal orbit achieved would be monitored by NASA's Deep Space Network. Once the proper disposal orbit has been verified, no additional monitoring is necessary. However, monitoring could be re-established in the future if required.

2.0 SUMMARY OF KEY FINDINGS AND CONCLUSIONS

2.1 KEY FINDINGS. The principal findings of the study are reported here as responses to questions which address the key study issues. The sequence of these findings is a logical progression, beginning with the choice of space disposal destination. Each issue is keyed to the appropriate section for reference if more detail is desired.

ARE THERE ALTERNATE DESTINATIONS THAT SHOW PROMISE? (Section 5.1)

A top-down study of 14 destinations showed none with less risk than the reference destination. Several destinations in the geolunar system offer the possibility of reduced cost with risk equivalent to the reference destination. Further analytical verification of orbit stability would be required to allow further consideration of these alternatives.

WHAT ARE THE ORBIT TRANSFER SYSTEM PERFORMANCE REQUIREMENTS? (Section 5.2)

Performance requirements for the chemical propellant mission (impulsive burns) include an injection delta-V (to the heliocentric transfer orbit) of 3.274 km/sec with a placement delta-V of 1.283 km/sec. Mission duration is about 165 days. If electric propulsion is used, the delta-V increases to 10.249 km/sec for the total mission due to gravity losses, and the mission duration increases to 545 days.

WHICH ORBIT TRANSFER VEHICLE OPTIONS ARE BEST CAPABLE OF PERFORMING THE SPACE DISPOSAL MISSION? (Section 5.3)

Orbit transfer system options were investigated which were compatible with dual launch missions (two launch vehicle payloads combined in low Earth orbit) and single launch missions. Cryogenic and storable liquid propellants and electric propulsion options were investigated. Three systems showed increased performance when compared to those used in the reference concept. Dual launch missions were best suited to a large two-stage system, with both injection and placement stages using cryogenic propellants. The injection stage is reused. A single-stage cryogenic propellant long life OTV used in the expendable mode

was the best single launch chemical propellant option. The optimum single launch electric propulsion vehicle was sized at 270 kW and operates in an expendable mode direct from low orbit to the destination.

WHICH LAUNCH SYSTEMS OFFER THE BEST COMBINATIONS OF COST, RISK, AND AVAILABILITY? (Section 5.4)

Evaluation of a spectrum of launch systems ranging from the existing space shuttle to projected heavy lift launch vehicles capable of orbiting 200 metric tons per launch led to selection of two vehicles as being the most cost effective. Single launch scenarios were served by the same uprated shuttle used in the reference concept; for dual launch missions, the uprated shuttle was teamed with a shuttle derivative cargo launch vehicle using liquid rocket boosters. This combination takes advantage of the orbiter's intact abort capability to minimize risk to the waste payload during an abort, but uses the efficient shuttle derivative to carry the heavier orbit transfer system. This combination offers risk equivalent to or lower than the reference uprated space transportation system (STS) while offering significant life cycle cost savings over the single launch option for a wide range of assumptions on DDT&E and recurring costs.

WHAT ARE THE REQUIREMENTS FOR PROTECTION OF THE WASTE PAYLOAD? (Section 5.5)

A review of previously established safety requirements and of reliability data relative to active escape (ejection) systems resulted in the addition of an orbiter crash condition to the previous blast overpressure, fragment penetration and thermal accident environments. The radiation shield criteria was reduced to 1 rem/hr at 1m from the 2 rem/hr level used in the reference concept to bring it into conformance with existing specifications for nuclear waste transportation system post-accident exposure levels.

WHICH WASTE PAYLOAD PROTECTION OPTIONS ARE MOST EFFECTIVE IN MINIMIZING MISSION RISK? (Section 5.5)

An evaluation of a wide range of containment and radiation shield material options, shield removal options, and ejection system options led to adoption of an integral (non-removable and hermetically sealed) radiation shield and

containment system fabricated from high strength steel and graphite for all options. This option satisfies radiation shielding and thermal requirements and provides maximum resistance to physical insults such as impact, fragment impingement and blast overpressure. The increased performance of selected launch ard orbit transfer systems allowed the required increase in waste payload mass without increasing tne cost of payload delivery to the 0.85 AU destination.

WHAT IS THE FACILITIES IMPACT AT KSC, AND WHAT ARE THE IMPLICATIONS OF USING ALTERNATE LAUNCH SITES? (Section 5.8)

The selected alternate space systems would approximately double the existing STS facilities at KSC. Additional facilities would be limited to the NPPF used for waste payload receiving and pre-launch processing. Evaluation of alternate sites found no significant performance or risk benefits. The primary driver for adopting ar alternate site would appear to be political in the context of an international space disposal scenario.

WHICH COMBINATIONS OF ORBIT TRANSFER, LAUNCH, WASTE PAYLOAD AND FLIGHT SUPPORT SYSTEMS OFFER THE BEST COMBINATIONS OF COST AND RISK? (Section 5.7)

The primary evaluation criteria for total system merit were (1) relative risk to be less than the Marshall Space Flight Center reference concept and (2) relative cost less than or equal to the Marshall Space Flight Center reference concept.

Four systems were selected as alternative systems possessing high merit. Two single launch solar electric options utilize high specific impulse to deliver 5405 kg of waste payload fully shielded to the selected 0.85 AU destination. Two dual launch chemical propellant concepts use economies of scale in launch vehicles and the increased specific impulse offered by long life cryogenic placement stages to deliver 10,150 kg of waste form fully shielded to the 0.85 AU destination.

Figure 2.1-1 provides a schematic illustration of key mission events for the chemical propellant, dual launch option. Key events include:

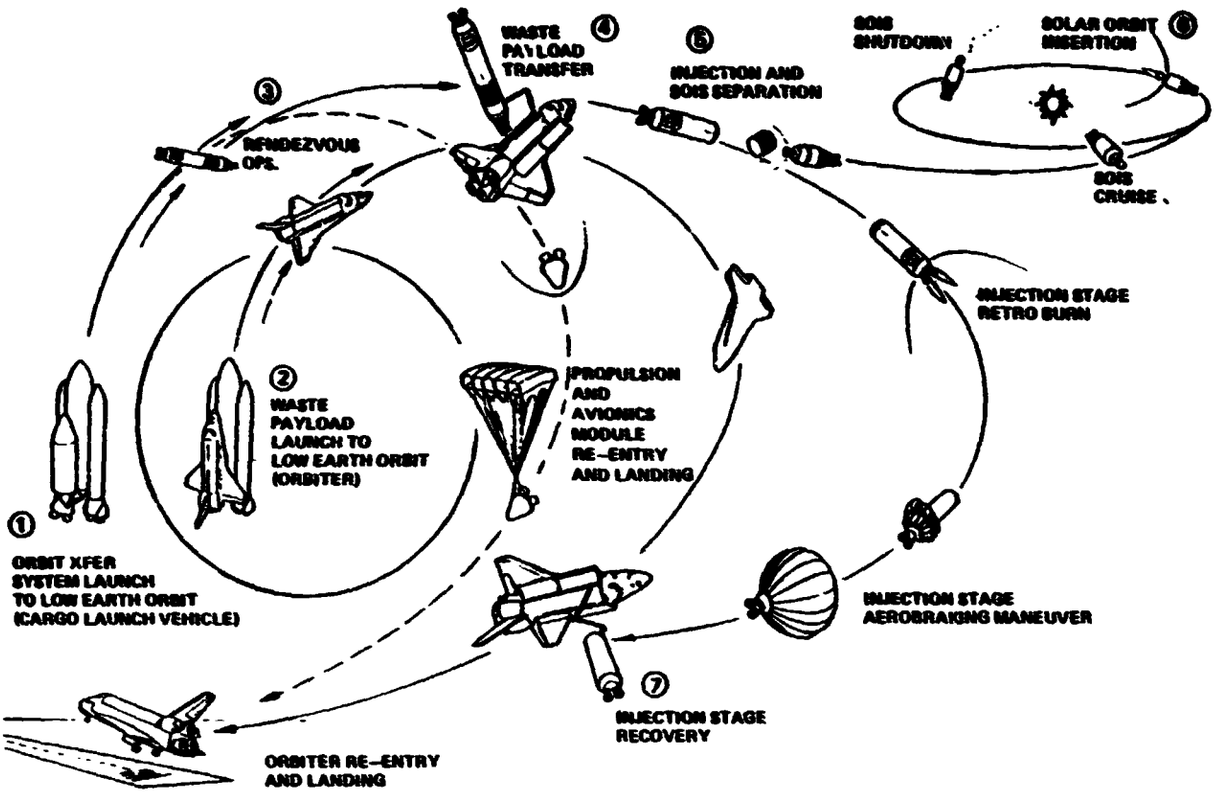

WASTE PAYLOAD TRANSFER
INJECTION AND SOIS SEPARATION
SOIS SHUTDOWN
SOLAR ORBIT INSERTION
RENDEZVOUS OPS.
SOIS CRUISE
INJECTION STAGE RETRO BURN
WASTE PAYLOAD LAUNCH TO LOW EARTH ORBIT (ORBITER)
PROPULSION AND AVIONICS MODULE RE-ENTRY AND LANDING
ORBIT XFER SYSTEM LAUNCH TO LOW EARTH ORBIT (CARGO LAUNCH VEHICLE)
INJECTION STAGE AEROBRAKING MANEUVER
INJECTION STAGE RECOVERY
ORBITER RE-ENTRY AND LANDING

Figure 2.1-1: Orbital Operations for the Chemical Propellant, Dual Launch Option

1) Launch of the cargo launch vehicle which places the two stage orbit transfer system in low Earth orbit (LEO).

2) Launch of the waste payload to LEO in the uprated space shuttle.

3) Rendezvous between the orbit transfer system and orbiter in LEO.

4) Transfer of the waste payload to the orbit transfer system from the flight support system (FSS) which supports it in the orbiter cargo bay. Subsequent to waste payload transfer, the orbiter waits in LEO for recovery of the first stage of the orbit transfer system.

5) Injection of the expendable solar orbit insertion stage (SOIS) into heliocentric transfer orbit by the recoverable first stage.

6) After a 165 day coast, the SOIS injects itself and the waste payload into the destination heliocentric orbit at 0.85 AU.

7) Recovery of the injection stage for reuse following a retroburn and aerobraking maneuver which inserts it into LEO.

Key events in the mission sequence for the selected single launch, electric propulsion option are illustrated schematically in Figure 2.1-2. The mission se-

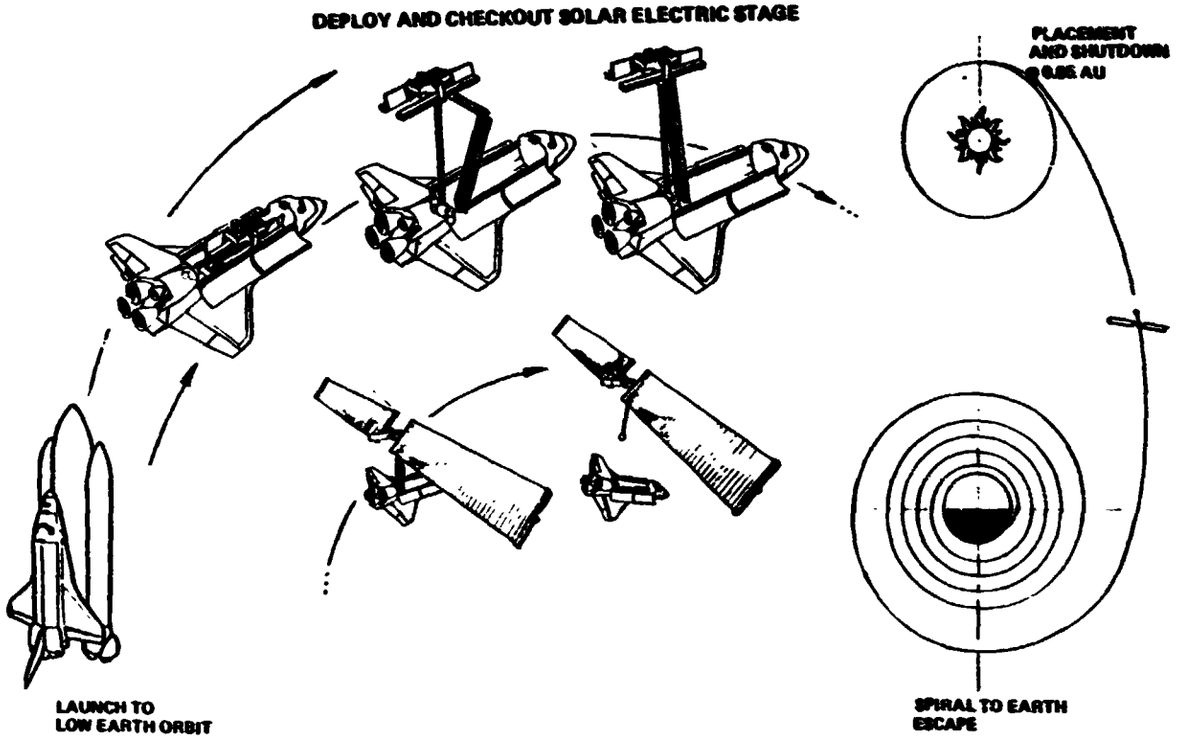

DEPLOY AND CHECKOUT SOLAR ELECTRIC STAGE
PLACEMENT AND SHUTDOWN @ 0.85 AU
LAUNCH TO LOW EARTH ORBIT
SPIRAL TO EARTH ESCAPE

Figure 2.1-2: Orbital Operations for Single Launch, Electric Propulsion Option

quence is significantly simpler than the dual launch option. Launch vehicle ascent operations are followed immediately by deployment and checkout of the orbit transfer system while still attached to the shuttle. At the completion of checkout, waste payload transfer is accomplished by unlatching the waste payload from the flight support system after which the orbiter backs away from the orbit transfer system and payload and proceeds immediately to recovery and landing. Following separation, the solar electric stage begins its injection spiral. Completion of the injection spiral leaves it on a two-third turn transfer spiral to the 0.85 AU destination. Placement is accomplished by shutdown of the solar electric propulsion system when 0.85 AU is achieved followed by verification of destination orbit parameters by ground control and the permanent shutdown of the solar electric stage, completing the deployment.

2.2 MAJOR CONCLUSIONS. **Four alternate space systems for disposal of nuclear waste possessing high merit have been identified. These systems offer substantial risk benefits when compared to the reference system at comparable or lower cost.**

The alternative concepts for space disposal take advantage of new systems to provide enhanced performance. All four concepts approximately triple the mass delivered to the reference 0.85 AU destination when compared to the reference concept at a comparable cost. These performance advances are made possible by:

1) Use of the shuttle derived cargo launch vehicle.

2) Economies of scale in large cryogenic propellant injection stages and cryogenic propellant placement stages.

3) Increased propulsion performance due to the higher specific impulse provided by cryogenic placement stages and electric propulsion for both injection and placement.

All four concepts use high performance to provide low risk. The key factor in risk reduction is the tripling of payload mass delivered which allows the full 5000 kg reference concept waste form to be shipped all the way to the destination fully shielded. This provides protection for the payload against all potential insults during all mission phases. Additional factors contributing to the reduction of mission risk include the ability to provide an ejectable waste payload protection system to provide protection from accidents to the launch system and the ability to ship the waste payload to low orbit in a payload bay which does not contain high energy or hypergolic orbit transfer system propellants.

The alternative space transportation concepts identified can be realized using existing technology. Mass and performance estimates have been calculated using characteristics of existing or demonstrated (prototype) hardware. Space transportation systems for the disposal of nuclear waste require systems development, not technology development.

Technology development is required for the waste payload. Significant waste payload technology areas requiring further investigation include containment and waste form fabrication. The ability of the waste payload to maintain containment of the waste form following terminal velocity impact requires verifcation by both analysis and test. Fabrication of cermet waste forms of the sizes considered in this study is not possible using existing technology. Further investigation of this area should not begin, however, until planned studies of alternate waste mixes/forms are complete.

Significant remaining problem areas include contingency rescue and post-burial meltdown. A contingency rescue involving location and docking with a non-cooperative target in heliocentric orbit is impossible using existing systems (see Section 4).

Meltdown following burial of the waste payload following a launch accident or unplanned re-entry could result in a melt through of the waste payload containment within 30 hours (see Volume 2, Section 6.3.5). As the possibility of burial cannot be totally eliminated, this factor may impose an upper limit on allowable waste payload thermal loading. The PW-4b waste mix investigated in this study certainly exceeds this limit.

3.0 STUDY LIMITATIONS

While this study was comprehensive in its treatment of most aspects of space systems, two key limitations remain.

1. The study was restricted to consideration of single waste mix and a single disposal rate requiring approximately 60 launches per year. Alternate waste forms requiring a lower launch rate for disposal could have fundamental impact on the selection of orbit transfer and launch systems.

2. A comprehensive payload response analysis for the waste payload system identified in this study is required for fragment impact and for terminal velocity impact.

Further details on the assumptions made in this study can be found in Volume 2, the Technical Report.

4.0 RECOMMENDATIONS FOR FURTHER STUDY

Specific recommendations for further study resulting from the space system study effort are summarized below. The primary additional study effort required is in the systems area. The fundamental viability of the concept is so closely tied to the structural integrity of the waste payload system that early investment in technology verification in this area is also recommended.

4.1 SYSTEM STUDY AREAS:

1. Further analysis should be conducted of space disposal destinations in the geolunar system. Efforts should be aimed at defining the best geolunar destination and validating its stability to the same level as the reference 0.85 AU heliocentric orbit destination. Validation would allow realization of the cost and risk benefits of the geolunar destination. Additional factors are:

   a. This destination option could be important if further studies of the contingency rescue mission find it infeasible or impractically expensive due to acquisition, tracking, or rendezvous/docking problems (see item 2).

   b. The reduced DDT&E and production costs due to deletion of the placement stage for geolunar destinations will be increasingly important if the launch rate is reduced as a result of adopting alternate, lower volume waste mixes.

      Use of geolunar system destinations would allow elimination of the placement stage with complete reuse of the injection stage, which could be an unmodified version of the OTV planned by NASA for operation in the 1990's. The resulting reduction in DDT&E and production costs will provide life cycle cost savings of increasing importance if alternate, lower volume waste mixes are adopted, allowing fewer flights for amortization of sunk costs.

2. A preliminary study of the contingency rescue mission in more detail than reported in past studies is required to identify concepts and define areas for further study more specifically. The goals of this effort should be to:

   a. Establish the quantitative risk benefits of maintaining the contingency rescue capability, as opposed to maintaining the normal rescue mission capability only.

   b. Establish the fundamental technical viability of contingency rescue in deep space.

   c. Estimate cost for implementation (particularly DDT&E costs).

   Accomplishment of these tasks will allow determination of whether contingency rescue is an enabling capability for space disposal and, if it is, will provide the basis for decisions on the level of emphasis to be applied to this area.

3. A system level study is required to determine the impact of alternative waste forms and mixes, primarily in the area of launch rate and its effect on selection of launch vehicles and orbit transfer systems. The systems treated in this study are optimized for a launch rate of 50 to 60 launches a year. Operations and launch vehicle costs dominate the life cycle costs, allowing a great deal of latitude for investment in advanced orbit transfer and launch systems without major life cycle cost impacts. Alternate waste mixes of reduced quantity would cut the cost base for system amortization, changing the criteria for system selection. Systems should be investigated for the entire range of potential waste quantities in sufficient depth to allow definition of "optimum" space systems for all reasonable waste forms/mixes.

4.2 TECHNOLOGY STUDY AREAS. An analysis of the reference integral shield waste payload system aimed at validating its ability to withstand terminal velocity impact should be conducted as the first part of a comprehensive payload

accident effects analysis for this concept. This effort would provide preliminary verification of the technical viability of the waste payload system and, by implication, the entire space disposal system. It would also be the first step in a more extensive effort aimed at the validation and qualification of the waste payload system.

5.0 SUMMARY OF TECHNICAL RESULTS

This section summarizes the significant technical results of the Boeing Aerospace Company study of space disposal of nuclear waste. This section is divided into eight subsections which correspond to the eight study tasks specified in the statement of work.

5.1 SPACE DISPOSAL DESTINATIONS. This section presents the results of our investigation of locations in space, or destinations, where nuclear waste payloads could be deposited for permanent isolation from the terrestrial biosphere. The investigation was aimed at defining a single destination to be used in definition of the space transportation systems required for the space disposal mission. Because of its pivotal role in setting space system performance requirements, this task was performed at the beginning of the study.

The primary issue was to determine if there was a safer or cheaper destination than the circular heliocentric orbit at 0.85 AU chosen as a reference destination in the 79-80 stuoy effort.

An initial screening which encompassed a wide range of destinations in the solar system as well as solar system escape orbits resulted in the selection of the 14 candidate destinations snown in Figure 5-1 for evaluation in depth. The selected candidate destinations were evaluated for cost related factors primarily based on performance, risk related factors including potential for return of waste material to the terrestrial biosphere, and other factors including implications for future use and retrievability.

A final screening was conducted in which each candidate destination was ranked in terms of these three criteria. Candidates with costs and risks higher than those of the reference helicentric orbit destination were rejected for further consideration as were those which evidenced a problem with contamination of destinations which could compromise potential future usage.

Six candidate destinations satisfied all three criteria for selection: both posigrade and retrograde high Earth orbits, orbits about the Earth-Moon Trojan points, lunar orbits, and heliocentric circular orbits at both 0.85 and 1.15 AU

LOCATION	TYPE	CATEGORY	CANDIDATE DESTINATION	REF. NO.	RATIONALE FOR SELECTION
GEOLUNAR SYSTEM	SURFACE	MOON	LUNAR SURFACE	1.	ONLY AVAILABLE
	ORBITS	HEO	HEO POSIGRADE	2.	LOWEST Δv
			HEO RETROGRADE	3.	ENHANCED STABILITY
		LIBRATION POINTS	L4 TROJAN ORBITS	4.	ENHANCED STABILITY
		LUNAR ORBITS	LUNAR ORBITS	5.	MINIMIZE RE-ENCOUNTER PROBABILITY
DEEP SPACE	SURFACE	PLANETS	VENUS IMPACT	6.	LOWEST Δv
			JUPITER ENTRY	7.	NO LONG TERM CONTAMINATION
		MOONS	LUNA	1.	LOWEST Δv (SEE GEOLUNAR)
		ASTEROID	APOLLO - AMOUR GROUP SOFT LANDING	8.	LOWEST Δv, TRUE TIMES
		SUN	SOLAR IMPACT	9.	ONLY AVAILABLE
	ORBITS	HELIOCENTRIC	EARTH ESCAPE : _IPTICAL	10.	LOWEST Δv FOR EHLIOCENTRIC
			0.85 AU CIRCULAR	11.	LOWEST Δv FOR VERIFIED STABILITY
			1.15 AU CIRCULAR	12.	SEE GEOLUNAR
		LIBRATION POINTS	SUN-EARTH "TADPOLE" ORBITS	13.	LOWEST Δv ; STABILITY VERIFIED
		SOLAR SYSTEM ESCAPE	ESCAPE HYPERBOLA	14.	LOW LONG TERM RISK

Figure 5-1: Candidate Destinations

radius. From these six, the 0.85 AU radius heliocentric orbit was selected as a reference for the study. It offers a combination of the lowest long term risk of any destination studied and excellent characterization from previous studies.

A high Earth orbit destination was recommended for evaluation as an alternate to the heliocentric orbit reference. A full evaluation of long term risk for this class of destination is not available, but potential risk could be comparable to the reference and some attractive cost benefits would result from factors such as lower performance requirements for orbit transfer systems. When evaluated, these cost benefits would indicate the potential payoff which could justify a later in-depth investigation of the concept's long term risk.

5.2 **MISSION AND OPERATIONS ANALYSIS.** This task was conducted in three parts. The first, analysis of delivery mission profiles, describes the analysis of the payload delivery to the 0.85 AU heliocentric orbit destinations selected in the destination task. The second part describes the analysis of mission operations for the selected missions, and the third the studies of mission control requirements for the space disposal mission selected.

DELIVERY MISSION ANALYSIS. Objectives of the delivery mission analysis effort were to provide mission profiles in terms of event sequences, time lines and performance requirements for the delivery mission. Key mission profile events defined for all chemical propulsion orbit transfer system options included injection, injection stage (OTV) recovery, coast and placement. A separate mission analysis was provided for the solar electric orbit transfer system options.

Injection Mission Profile. The injection mission profile developed is illustrated in Figure 5.2-1. The profile shown is used with a recoverable OTV, but the

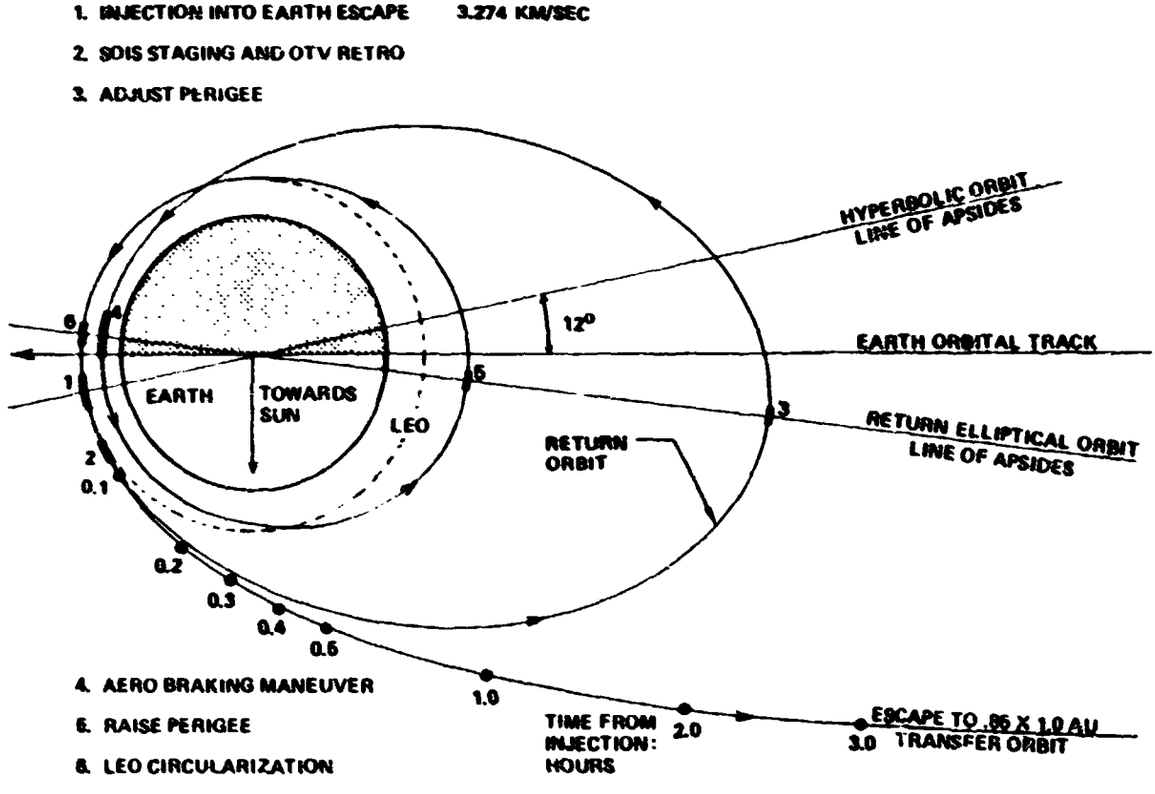


Figure 5.2-1: Injection Mission Profile Schematic

injection hyperbolic escape orbit is equally applicable to expendable vehicle options. Key events in the injection mission profile are numbered from 1 through 6 in the illustration.

Injection Stage Recovery. Recovery of the injection stage following solar orbit insertion stage injection into heliocentric transfer orbit was an important contributor to reducing the cost of the disposal mission. A review of past references was unable to uncover definitive data on optimization of OTV recovery trajectories. Accordingly a study was conducted to define optimum recovery modes for both all propulsive and aerobraked OTV recovery options. Results of the study showed that the return mission period may be chosen any time between 40 and 100 hours with little overall penalty. The duration for the selected aerobraked return chosen was 62 hours which is very close to optimum and which matches the capability of the baseline orbit transfer vehicle studied in the OTV Phase A Concept Definition Study.

Placement Mission Profile. Figure 5.2-2 schematically illustrates the placement mission profile. Primary events following injection include an optional

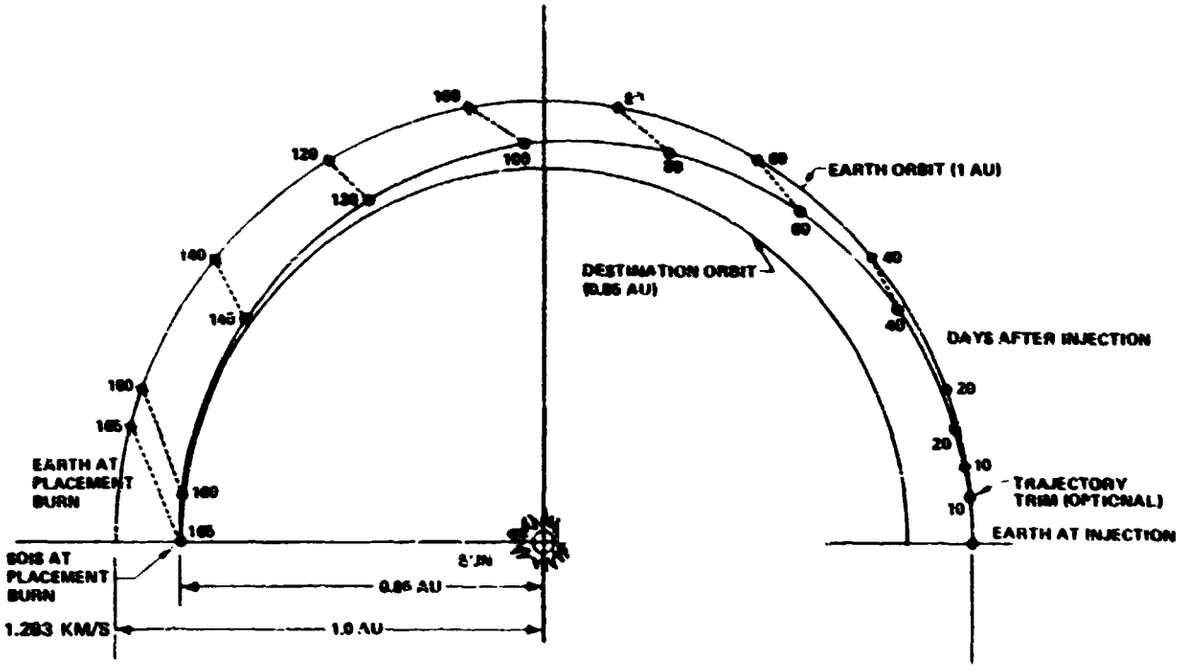


Figure 5.2-2. Placement Mission Profile Schematic

trajectory trim maneuver at about injection plus 10 days to correct for injection inaccuracies. The subsequent 155 day coast to periapsis of the transfer orbit is followed by orientation and the placement burn of 1.283 km/sec.

Solar Electric Stage Mission Profile. The mission profile developed for the solar electric stage is illustrated in Figure 5.2-3. Key events include a

LEO ORBIT: 100NM @ 28.5°	
TIME TO 0.85AU	545 DAYS
ΔV TO ESCAPE	7.728 Km/SEC
ΔV ESCAPE → 0.85AU	2.521 Km/SEC
	10.249 Km/SEC

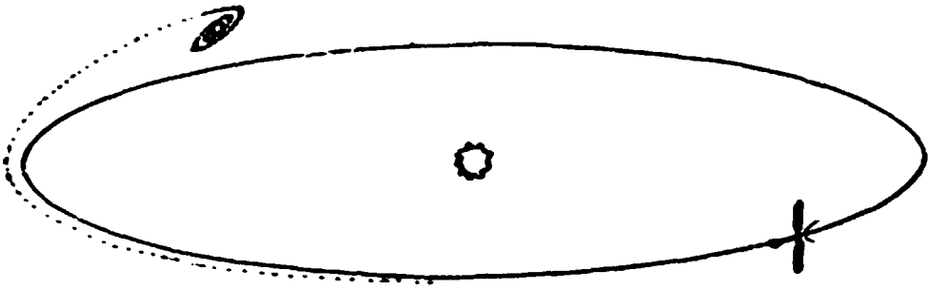


Figure 5.2-3: Solar Electric Stage Mission Profile Schematic

multiturn spiral from lower Earth orbit toward the escape injection involving a delta-V of approximately 7.7 km/sec over a period of about 1 year. This is followed by a two-thirds turn spiral into the heliocentric orbit destination involving a delta-V of about 2.2 km/sec applied over a duration of 180 days. Total mission duration is about 545 days.

RESCUE MISSION ANALYSIS. Rescue mission analysis was accomplished for high Earth orbit and deep space rescue missions to provide definition of: 1) abort modes, 2) trajectories for rescue, and 3) the performance requirements in terms of delta-V and duration required to implement the rescue missions.

Definition of Abort Options. Abort mode studies allowed definition of a logical abort strategy, which required consideration of only two basic rescue mis-

sion profiles: (1) deep space rescue with the target vehicle in a 0.86 x 1.0 AU transfer orbit ellipse, and (2) high earth orbit rescue with the target vehicle in a circular or slightly elliptical holding orbit at an orbital altitude between 500 and 40,000 km.

Rescue Mission Trajectories and Performance Estimates. Trajectories were generated for rescue missions involving payloads stranded in Earth orbit or deep space. Performance requirements for Earth orbit rescue required delta-V's of about 3.2 to 3.5 km/sec to reach the stranded payload; subsequent to waste payload transfer to the rescue vehicle, injection takes approximately 2.0 km/sec, followed by a standard placement.

Performance requirements established for deep-space rescue missions are bounded by the case of total SOIS failure. For a typical three-impulse transfer, the injection delta-V to ~ 0.85 AU perihelion is in the range of 3.5 km/sec, with two intercept delta-V's of approximately 1.2 km/sec each used for rendezvous with the target at the target's second perihelion, followed by a final 1.18 km/sec placement burn.

OPERATIONS ANALYSIS. The objective of the operations analysis task was to define system operations to the level required for system design support and to provide standards for qualitative differentiation between alternative concepts in the areas of mission complexity and risk evaluation.

The study resulted in the comprehensive top level operations flow illustrated in Figure 5.2-4. This flow encompasses 1) launch vehicle options using one or two launch vehicles per mission, 2) orbit transfer system staging and reuse options, 3) shield removal options. This top level flow was used as a basis for definition of key lower level operations flows as required. Detailed definition was provided for OTV/SOIS separation to allow evaluation of separation delays on orbit transfer vehicle return trajectory delta-V requirements. Data were derived from similar timelines defined for the Boeing/USAF IUS.

MISSION CONTROL. Mission control issues for the launch vehicles, orbit transfer vehicles, and solar electric stage have been well defined in ongoing studies of the space transportation system, inertial upper stage, and solar

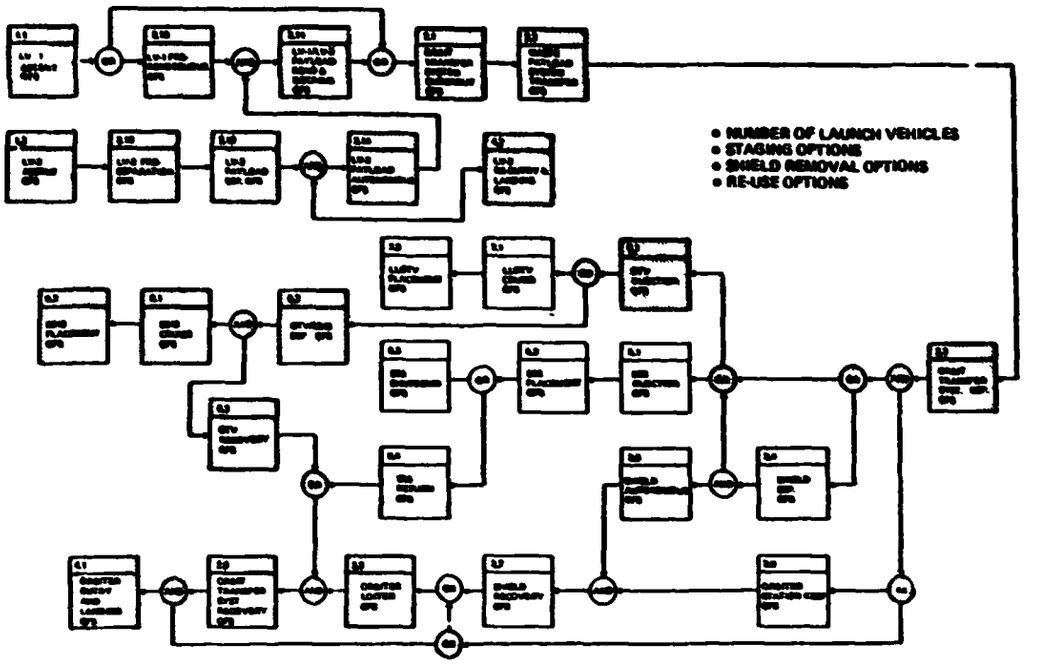


Figure 5.2-4: Space System Composite Operations Flow

electric propulsion systems. Control requirements for the SOIS however, remain largely undefined.

The available data base on vehicles such as Burner 2, IUS, and the Pioneer Venus Orbiter, along with consideration of SOIS functional requirements, indicated the potential for a significant simplification of SOIS avionics when compared to the avionics carried by the OTV. Accordingly, the thrust of this task became one of defining minimum requirements for the SOIS control functions for use in Task 6 vehicle definition efforts.

Functional Requirements. Functional requirements were defined for both three-axis and spin stabilized SOIS configurations for separation, cruise and placement operations. Primary requirements defined for both control modes include post-separation orientation verification, maintainance of cruise orientation, providing for contingency trajectory trim, and pitchover and orientation control during placement.

SOIS Control. Key features of the spin stabilized SOIS control system include attitude control using a combination of eight axial thrusters and four radial thrusters, with vehicle attitude determined by star and sun sensors. The Earth sensor serves for placement burn attitude verification. Features of the three -

axis stabilized SOIS control concept include attitude control provided by four thruster clusters which provide redundant thrusters for control of vehicle pitch, roll, and yaw. Vehicle attitude is determined by redundant sun sensors and star sensors. One of the star sensors is used as an Earth sensor for placement burn orientation verification.

5.3 **ORBIT TRANSFER SYSTEMS.** The objective in this task was to trade a full range of orbit transfer system options (propulsion, staging, reuse) in a systematic manner to define the optimum orbit transfer system for the space disposal mission. The approach chosen was to use the extensive existing data base on IUS, SEPS, and OTV as the basis for estimates of mass and performance. Trades were conducted in four primary areas: 1) assessment of propulsion systems for application, 2) identification of candidate systems and options, 3) definition of candidate vehicles, and 4) comparison of total system performance.

ASSESS PROPULSION SYSTEMS FOR APPLICATION. An initial screening of propulsion system candidates resulted in selection of cryogenic liquid, storable liquid, solid and solar electric propulsion for preliminary evaluation. Other concepts such as nuclear electric propulsion or laser propulsion were rejected as being difficult to characterize (particularly in cost) and of doubtful availability for a 1995 IOC. With candidate systems selected, a preliminary evaluation was conducted to assess propulsion systems for application to different mission phases.

Solid propellant rockets were rejected for further consideration. Solid rockets showed no advantage in specific impulse over storable liquid systems and are inflexible in terms of installation. Storable liquid propulsion systems were assigned to the placement role only. Their reduced specific impulse (300-310 sec compared to 460 sec) relative to cryogenic propellants for the injection mission led to their rejection for this relatively high delta-V application. Cryogenic liquid propellants and electric propulsion were determined to be suitable for all mission phases.

IDENTIFICATION OF ORBIT TRANSFER SYSTEM OPTIONS. A combination of the selected propulsion options by mission phase with the remaining options in the areas of staging and reuse provided definition of the entire range of orbit transfer system options for space disposal of nuclear waste. The results of this combination are illustrated in matrix form in Figure 5.3-1. Illustrated are combinations of options in the areas of staging, re-use, and propulsion which yield a total of 13 possible orbit transfer system options for characterization (option SES-3 was confined to use of an aerobraked injection stage for simplicity).

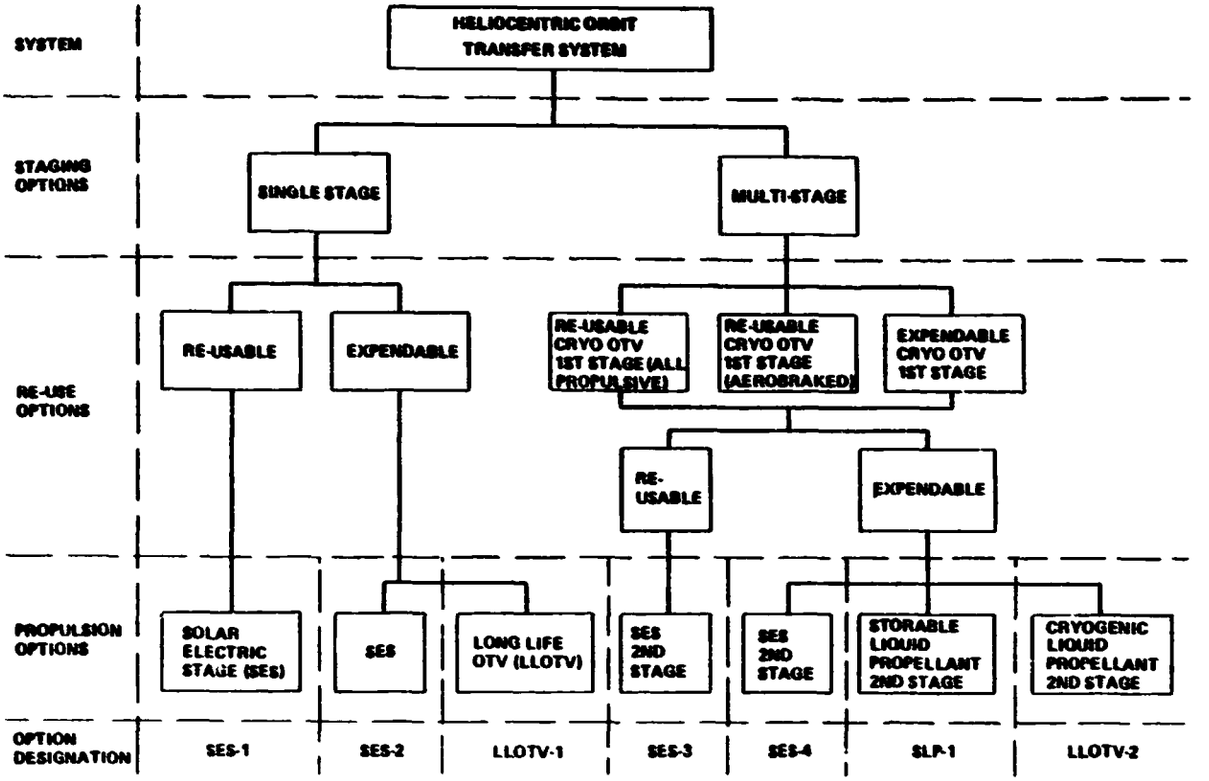


Figure 5.3-1: Identification of Candidate Orbit Transfer System for Space Disposal

DEFINE CANDIDATE VEHICLES. The vehicles defined for the different orbit transfer system options are illustrated in Figure 5.3-2. As the first step in determining the performance for the range of orbit transfer system options, point designs and parametric mass relationships were developed for the candidate vehicles illustrated. Point designs were prepared for: LOX/LH_2 cryogenic propellant vehicles, storable liquid propellant vehicles, and solar electric propulsion vehicles.

Cryogenic Propellant Vehicles. For the LOX/LH_2 stages the initial OTV defined in the Boeing/MSFC OTV Phase A Concept Definition Study (Contract NAS8-33532) was used as a reference point design. It is very close to the optimum size for the single launch LO_2/LH_2 injection stage. Point designs for all other LO_2/LH_2 vehicles were developed as variations from this baseline. The emphasis was on analyzing and defining those areas of the LO_2/LH_2 vehicles that were different from the Phase A baseline.

Storable Propellant Vehicles. The storable propellant placement stage was configured by combining existing engines (from the STS reaction control system)

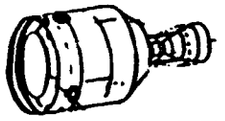
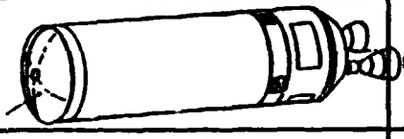
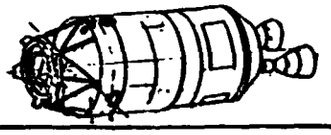
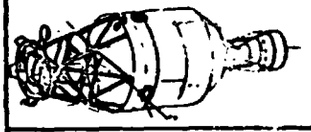
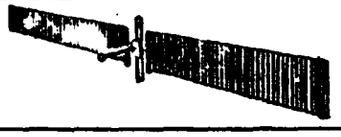
CONFIGURATION	USAGE	ROPELLANT	JOINT DESIGN CHARACTERISTICS (Kg)			DIMENSIONS
			MASS AT BURNOUT	PROPELLANT MASS	MASS AT STARTBURN	
	PLACEMENT (SOIS)	STORABLE N_2O_4 N_2H_4	1091	3316	4407	4.7 M DIA X 1.52 M LONG
	INJECTION	CRYO LOX LH_2	2931	21,100	24,811	4.7 M DIA X 8.36 M LONG
	INJECTION	CRYO LOX LH_2	5,529	60,135	66,820	4.7 M DIA X 17.7 M LONG
	INJECTION + PLACEMENT (LLOTV)	CRYO LOX LH_2	3,919	28,122	32,735	4.7 M DIA X 10.9 M LONG
	PLACEMENT (SOIS)	CRYO LOX LH_2	1,823	12,383	14,481	4.7 M DIA X 7.7 M LONG
	INJECTION + PLACEMENT	ARGON	4,770	11,225	16,332	145 M X 50 M (ARRAY)

Figure 5.3-2: Candidate Vehicles and Characteristics

and avionics (from the IUS third stage) with an in-house design for structure and electrical power. The extreme simplicity of the vehicle allowed a reasonable level of definition to be achieved within the constraints of the study.

Solar Electric Propulsion Vehicles. Definition of the solar electric stage drew heavily on the BAC SEPS* study effort and the previous Future Space Transportation Systems Analysis Study conducted by BAC for NASA/JSC (Contract NAS9-14323). A vehicle configuration developed for the space disposal mission in the JSC study was updated by using SEPS derived estimates for the key propulsion areas of arrays, power processors and thrusters. Structure, thermal, and propellant storage masses were scaled. Other subsystems such as avionics and RCS were synthesized using components derived from the IUS and OTV programs.

* Alternate System Design Concept Study for the Solar Electric Propulsion System (NAS8-33753)

Characterize Candidate System Performance. The vehicles defined in the previous task were used in the characterization of the candidate orbit transfer systems shown in Figure 5.3-1. Trending curves for mass at burnout were generated from the point design mass statements described in the previous section. Points selected from these curves were used for the performance analysis. The Payload and Sequential Mass Calculation (PSMC) program was then used to determine payload capabilities of the candidate orbit transfer systems as a function of mass at startburn.

SELECT CANDIDATE ORBIT TRANSFER SYSTEMS. The parametric performance characterization of orbit transfer systems provided the basis for performance comparisons leading to selection of four orbit transfer system options for consideration in Task 8. Comparisons were carried out in two steps for chemical orbit transfer systems. In the first, six candidate two-stage orbit transfer systems were compared to select the best two stage system. In the second step the best two stage systems were compared to single stage systems. The comparison is illustrated in Figure 5.3-3.

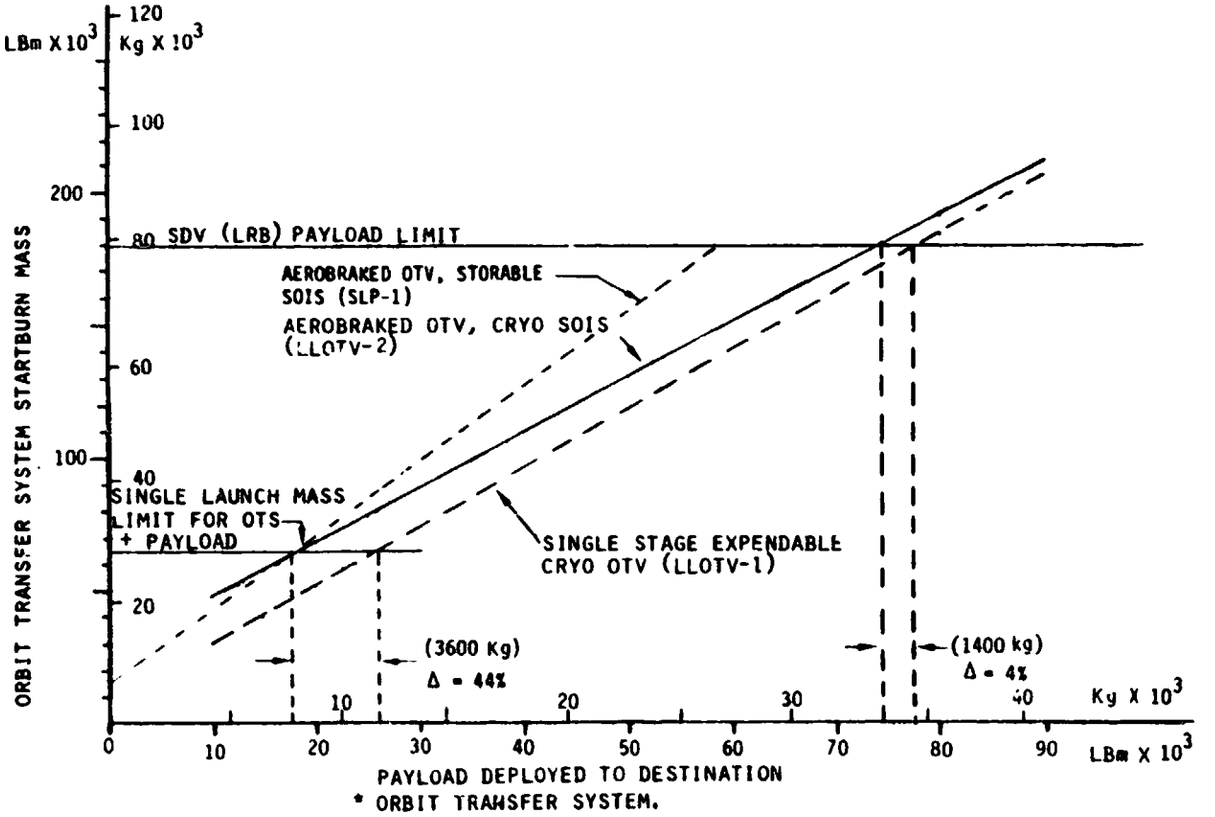


Figure 5.3-3: Parametric Performance Comparison of Staging Options

Comparisons were made for combined orbit transfer system and payload masses of about 82,000 kg, typical of dual launch options where the waste payload is carried up in one launch and the orbit transfer system in a separate launch; and for orbit transfer system and payload masses of about 23,000 kg, typical of single launch options with waste payload and orbit transfer system carried to low orbit in the same launch. Primary conclusions of the parametric performance comparison of chemical propellant systems include:

1. The maximum performance dual launch option is the aerobraked injection stage used with a cryogenic propellant SOIS.

2. A maximum performance single launch option is a single stage cryogenic propellant long-life OTV (LLOTV).

No performance trades were carried out for electric vehicles as only one option was selected.

SELECTED OPTION DEFINITION. Final orbit transfer system selection for consideration in Task 8 was based on performance and launch vehicle compatibility. Three options compatible with single launch missions were chosen and are illustrated in Figure 5.3-4.

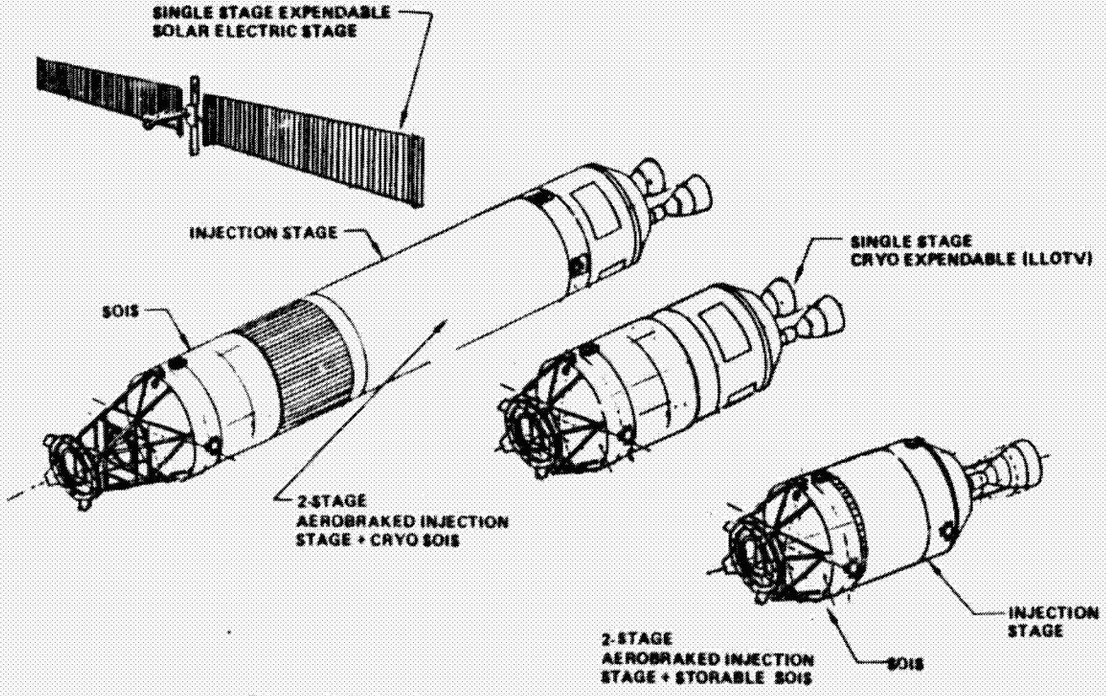


Figure 5.3-4: Selected Orbit Transfer Systems

1. The single stage cryogenic expendable LLOTV is the highest performance single launch chemical propellant option.

2. The two stage system using a storable propellant SOIS and an aerobraked injection stage is inferior in performance to the cryogenic propellant LLOTV but is closest to the reference system and was carried as a standard for evaluation of alternate systems.

3. The solar electric expendable stage has performance superior to the best chemical stage for the single launch option but is sufficiently different in cost to require cost comparisons at the total system level in Task 8.

The sole dual launch option considered is also illustrated in Figure 5.3-4. It uses a large aerobraked recoverable injection stage and an expendable cryogenic propellant SOIS. This combination offers the best overall performance in this size range.

RESCUE SYSTEM DEFINITION. Definition of the rescue system was required to allow calculation of rescue mission capabilities of the basic orbit transfer system delivery vehicles. Study of the MSFC reference system indicated potential mission cost savings in deletion of rescue-peculiar hardware from the standard SOIS. The study of rescue mission requirements conducted in Task 3 indicated that the only SOIS rescue-peculiar component required on every vehicle was a rendezvous radar transponder. Additional rescue provisions consisted of providing for a 3-year minimum life for the beacon transponder and attitude control system. This provided minimum cost for most missions but left the basic SOIS capabilities short of rescue mission requirements. The approach evolved was to kit the basic SOIS to provide a rescue vehicle with required capabilities.

Rescue System Requirements. Rescue mission performance requirements for the nominal rescue mission are described in Section 5.2. Additional requirements derived from analysis of rescue mission profiles include navigation to within terminal acquisition range of the target vehicle and accomplishment of terminal rendezvous and docking. Payload transfer to the rescue vehicle requires docking provisions on both the rescue vehicle and the vehicle to be rescued and effectors to accomplish payload transfer. The final requirement is to orient the SOIS for the placement maneuver and initiate SOIS autonomous operations.

Chemical Propellant Rescue System. The space system evolved to meet the rescue requirements specified in the previous paragraph consists of a dedicated rescue vehicle plus a standard orbit transfer system as used in the waste payload delivery mission. The rescue vehicle is assembled from a standard SOIS and a rescue kit, and is illustrated along with a summary mass statement in Figure 5.3-5.

	MASS, kg
STANDARD SOIS	14461 kg
–MAIN PROPULSION	
–ACS THRUSTERS	
–STAR SENSORS	
–PLUS–	
RESCUE KIT	1682
–TARGET ACQUISITION	
–RENDEZVOUS AND DOCKING	
–NAVIGATION	
–GROUND MONITORING	
–ADDED RCS PROPELLANT	
–ADDED ELECTRICAL POWER	
–JETTISONABLE	
ACTIVE DOCKING RING	95
–IMPACT ATTENUATION	
PAYLOAD TRANSFER MECH.	23
AFT SUNSHADE	78
–THERMAL CONTROL	
–SUPPLEMENTARY POWER	
–SUN SENSORS	
RESCUE VEHICLE GROSS	16239

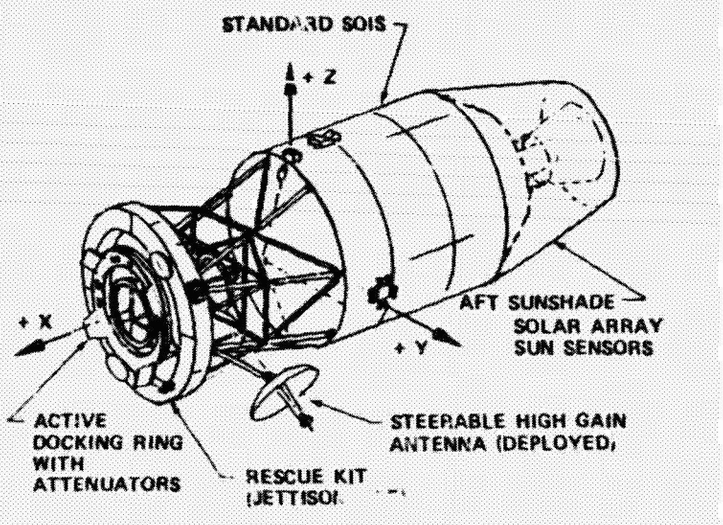


Figure 5.3-5: Rescue Vehicle Configuration and Summary Mass Statement

The rescue kit includes a reaction control system, propellant tankage, communication subsystem, a redundant inertial measurement unit, rendezvous radar electronics, computers and closed circuit television electronics unit which includes a high data rate RF subsystem and a deployable high gain antenna. Equipment is mounted in an equipment support ring which provides structural support and thermal control. A 3.1 meter diameter hole in the center of the equipment section provides for transfer of the waste payload.

The rescue kit is strut mounted to the SOIS payload support structure. Additional components include an aft sun shade which mounts a solar array and sun sensor used for vehicle attitude control and power during coast in the pursuit mode. At the conclusion of rescue operations the rescue kit and sun shade are jettisoned, converting the rescue vehicle back to a standard SOIS.

The mass of the standard SOIS is increased from approximately 14,400 kg to about 16,200 kg by the addition of rescue provisions. The largest increment is provided by the 1582 kg rescue kit. The active docking ring weighs 95 kg, mechanization for payload transfer an additional 23 kg, and the aft sunshade with its thermal control, supplementary power, and Sun sensors weighs 78 kg.

Rescue Orbit Transfer System. The rescue orbit transfer system consists of a pursuit configuration which carries the rescue vehicle to the target after injection and the injection configuration which injects the pursuit configuration to its initial transfer orbit.

The pursuit configuration consists of the rescue vehicle docked to the waste payload support structure of a standard SOIS. After injection and between maneuvers the pursuit configuration flies with the rescue vehicle sunshade pointed at the Sun, allowing the rescue vehicle to shade the standard SOIS. All the control functions are provided by the rescue vehicle; the standard SOIS serves as a propulsion module only. The same system is used with a slightly different delta-V split for the Earth orbit rescue.

The injection configuration of the rescue orbit transfer system is illustrated in Figure 5.3-6. The injection configuration is assembled on orbit using two launches. The first launch delivers a standard orbit transfer system to LEO using a shuttle derived cargo launch vehicle. The rescue vehicle is carried up in the second launch using the uprated shuttle orbiter. Injection of the pursuit configuration to its initial transfer orbit is accomplished by the large cryogenic injection stage which uses an aerobraking maneuver to return to low orbit for recovery by the orbiter.

Electric Propulsion Rescue Systems. Due to the relative lack of maneuverability of the electric orbit transfer system, the rescue kit for the electric vehicle is fitted with thrusters and used as a free flier capable of leaving the rescue electric vehicle to rendezvous with the failed vehicle. The waste payload is then transfered from the failed vehicle to the free flier rescue kit which undocks from the failed vehicle and returns and docks with the rescue vehicle. Following a hard dock which attaches the waste payload to the rescue vehicle, the free flier rescue kit is jettisoned.

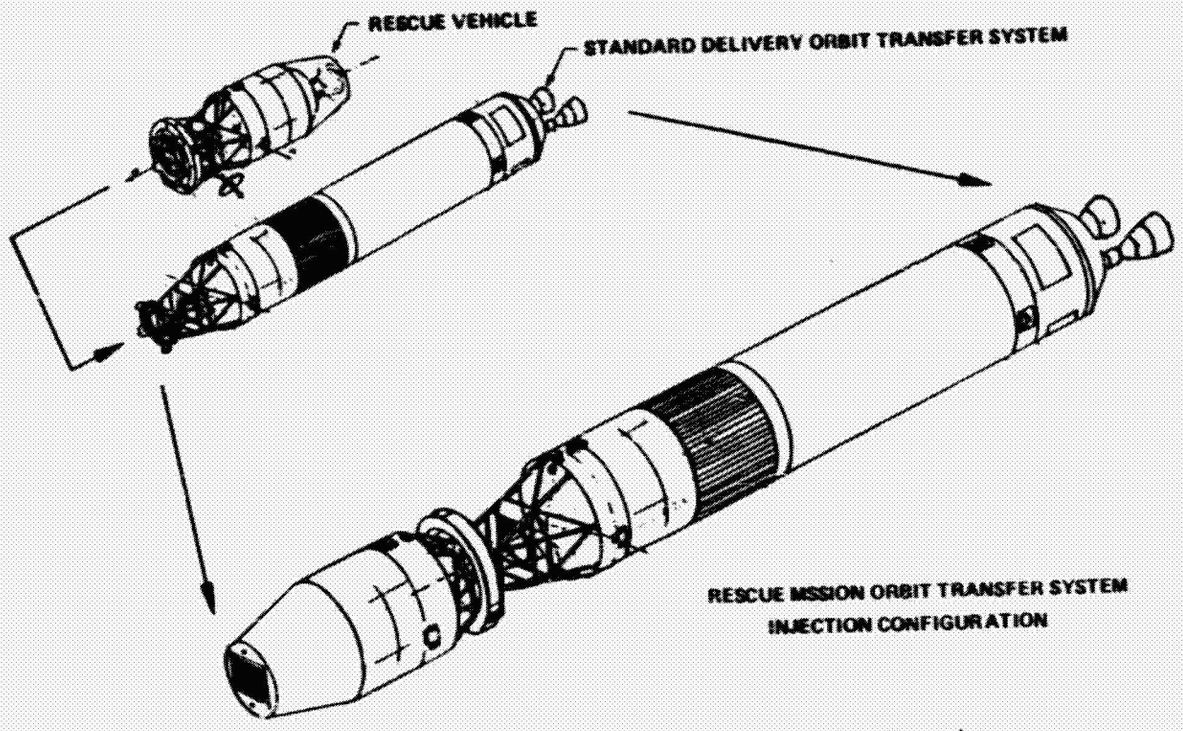


Figure 5.3-6: Rescue Orbit Transfer System: Injection Configuration

5.4 LAUNCH VEHICLE SYSTEMS. The primary objective of the launch vehicle systems task was to trade a full range of launch vehicle options in a systematic manner to define the optimum launch system for the space disposal option. A secondary objective included definition of unique requirements imposed on launch vehicle systems by the space disposal mission.

A preliminary screening of launch vehicle options resulted in selection of solid and liquid rocket boosted versions of the space shuttle orbiter and shuttle derived heavy lift vehicles as candidate configurations. Applicable references were reviewed to define key characteristics of the candidate vehicles in the areas of performance and cost. The candidate concepts were then traded in the areas of risk and cost to select the best system for the space disposal mission.

LAUNCH SYSTEM SCREENING. A survey of the available data on launch vehicles and past studies of space disposal of nuclear waste identified six categories of launch system candidates illustrated in Figure 5.4-1. A preliminary screening of the selected concepts eliminated the HLLV and SSTO concepts from further consideration on the basis of high capital costs and relative lack of definition.

(1) REFERENCE (29,500 KG PAYLOAD) SPACE TRANSPORTATION SYSTEM (STS)
(2) UPRATED (45,000 KG PAYLOAD) STS
(3) SHUTTLE DERIVATIVE CARGO LAUNCH VEHICLE USING SOLID ROCKET BOOSTERS [SDV(SRB)]
(4) SHUTTLE DERIVATIVE CARGO LAUNCH VEHICLE USING LIQUID ROCKET BOOSTERS [SDV(LRB)]
(5) HEAVY LIFT LAUNCH VEHICLE (HLLV) CONCEPTS
(6) SINGLE STAGE TO ORBIT (SSTO) LAUNCH VEHICLE CONCEPTS

Figure 5.4-1: Candidate Launch Systems

CANDIDATE LAUNCH SYSTEM DESCRIPTION. The candidate launch vehicles resulting from the preliminary screening are illustrated along with key characteristics in the areas of risk, cost, and performance in Figure 5.4-2.

Risk is expressed in terms of whether or not the vehicle possesses an intact abort capability. Winged orbiters in the event of a malfunction are able in

KEY CHARACTERISTICS / LAUNCH VEHICLE	STANDARD SHUTTLE	UPRATED SHUTTLE	SHUTTLE DERIVATIVE (SOLID ROCKET BOOSTER)	SHUTTLE DERIVATIVE (LIQUID ROCKET BOOSTER)
CONFIGURATION				
MAJOR ELEMENTS	• ORBITER • EXTERNAL TANK • SOLID ROCKET BOOSTER (2)	• ORBITER • EXTERNAL TANK • LIQUID ROCKET BOOSTER (2)	• EXPENDABLE CARGO SHROUD • RECOVERABLE PROPULSION AND AVIONICS MODULE • SOLID ROCKET BOOSTER (2)	• EXPENDABLE CARGO SHROUD • RECOVERABLE PROPULSION AND AVIONICS MODULE • LIQUID ROCKET BOOSTER (2)
APPLICATION	CREW AND CARGO	CREW AND CARGO	CARGO	CARGO
CARGO BAY DIA x LENGTH	4.57 x 18.3	4.57 x 18.3	7 x 24	7 x 24
PAYLOAD TO LEO Kg @ 370 Km	29,500 (5)	47,000	87,700	94,000
DDT&E	-	2.0 B	1.2 B	1.0 B (4)
PPOD COST (3) (4)	0	400 M	335 M	700 M
COST/FLT (3)	28.6 M	27.0 M	22.0 M	18.7 M

ALL COST IN 1980 DOLLARS

(1) ORBITERS NOT INCLUDED

(2) FOR FLIGHT RATE 23/YR

(3) WHERE APPROP INCLUDES ET AND P/A MOD

(4) WHEN DEVELOPED IN ASSOCIATION WITH DEVELOPMENT OF UPRATED SHUTTLE.

(5) TO 370 Km

Figure 5.4-2: Candidate Launch Vehicle Characteristics

most cases to jettison the external tank and glide back for a landing at the launch site or at an alternate field. Shuttle derivative vehicles do not possess this capability.

Performance is expressed in terms of payload bay size and the payload that the candidate vehicle can lift to a 28.5 deg inclination orbit at an altitude of 160 nmi. Cost is expressed in terms of the design, development, test, and engineering required to implement the candidate, the production cost per unit, and the cost per flight.

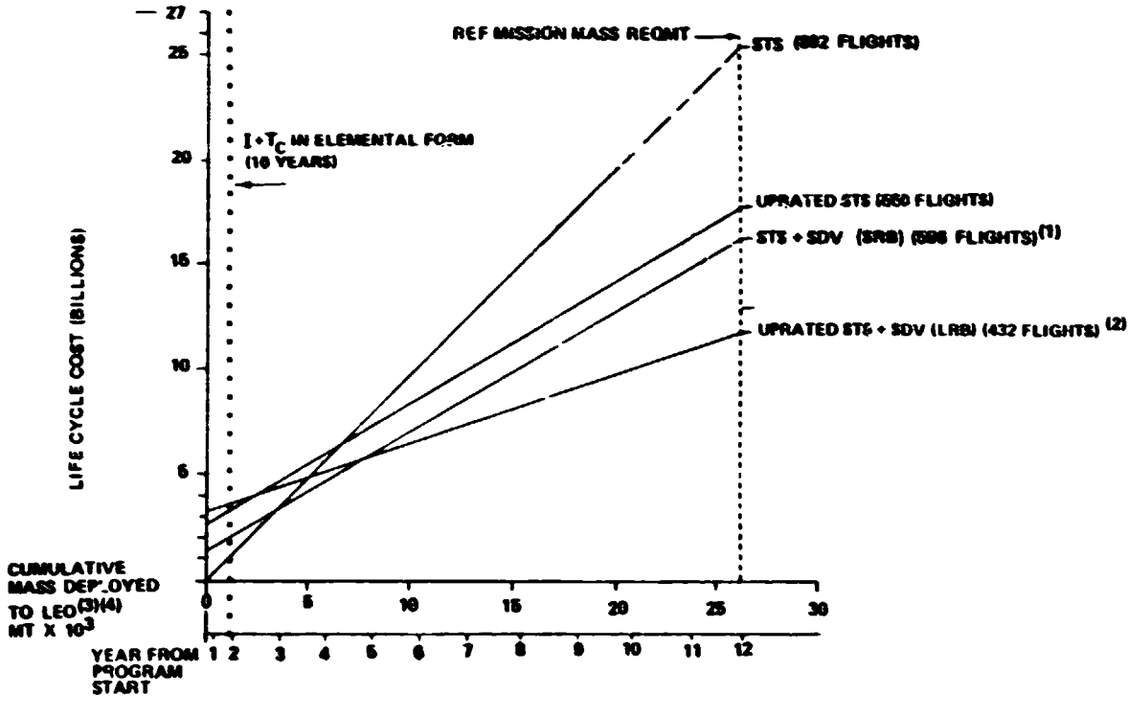
LAUNCH SYSTEM SELECTION. Launch systems selection was accomplished in two steps: in the first step, candidate launch systems were ranked for risk; the second step compared launch system concepts on the basis of life cycle cost.

Risk Assessment. Risk characteristics of the candidate launch systems are dominated by the lack of abort capability for the shuttle derived cargo launch vehicles. The logical conclusion is that the orbiter will always provide lower risk for the waste payload. The significantly lower cost per pound of the shut-

tle-derived cargo launch vehicle can still be used as part of a dual launch scenario with the waste payload carried in the winged orbiter and the orbit transfer system carried to low Earth orbit by the shuttle-derived vehicle. This concept preserves the risk benefits of the winged orbiter but uses the increased c st efficiency of the shuttle-derived vehicle to lift the heavier part of the space system.

Cost Assessment. As a result of the risk assessment, four space systems were carried into the cost assessment. Two single launch options include the reference space shuttle and the uprated space shuttle using liquid rocket boosters. Two dual launch options team the reference space shuttle with the solid rocket booster version shuttle derived cargo launch vehicle and team the uprated space shuttle with the liquid rocket booster version of the shuttle derived vehicle.

Candidate system life cycle launch system costs for the 10 year reference mission were compared by calculating the number of flights required for each candidate system to lift the full mission cumulative mass to low Earth orbit. Figure 5.4-3 compares launch system life cycle cost and shows some of the key assumptions used in their calculation. Launch costs for the four candidate systems



(1) INCLUDES 298 ORBITER AND 298 SDV FLIGHTS TO SUPPORT 298 MISSIONS
(2) INCLUDES 216 ORBITER AND 216 SDV FLIGHTS TO SUPPORT 216 MISSIONS
(3) BASED ON MSFC REF. SPACE SYSTEM CONCEPT
(4) INCLUDES WASTE PAYLOAD AND PROTECTIVE SYSTEM, FLIGHT SUPPORT SYSTEM, ORBIT TRANSFER SYSTEM

Figure 5.4 3: Life Cycle Cost Comparison for Candidate Launch Vehicles

are represented by the four lines running from left to right. The slope intercept represents DDT&E for initial deployment of the launch system; values range from 0 for the reference shuttle to about 3.2 billion for the uprated shuttle teamed with the liquid rocket booster version of the shuttle derived cargo launch vehicle. The slope of each line is proportional to the cost per flight.

Despite the highest initial DDT&E costs, the uprated shuttle/SDV team shows the lowest life cycle cost by a significant amount. This finding was tested by a series of sensitivity studies which indicated that this system shows the lowest life cycle cost for all cases studied. This advantage increases with increasing payload models and is maintained with smaller payload models down to about 20% of the reference mission studied. This concept was selected as the reference dual launch/launch system configuration for evaluation in Task 8.

The uprated shuttle using liquid rocket boosters was selected as the candidate for single launch vehicle mission scenarios in Task 8. This system shows a life cycle cost about two-thirds that of the reference space shuttle, which uses solid rocket boosters, and is identical to the launch system used in the reference concept. The liquid rocket boosters also significantly reduce the risk to the waste payload incurred in launch pad accidents by eliminating the possibility of encountering the extreme thermal environment associated with potential solid rocket booster fires.

LAUNCH VEHICLE UNIQUE REQUIREMENTS. Launch vehicle unique requirements identified for launch vehicles carrying a waste payload ejection system (see Section 5.5) are listed in Figure 5.4-4. Specific requirements identified for orbiters carrying waste payload systems not requiring ejection are limited to modifications required to allow landing with a maximum weight payload.

- JETTISONABLE/QUICK CYCLE PAYLOAD BAY DOORS
  - -FOR WASTE PAYLOAD JETTISON
  - -MAY OCCUPY ONLY PART OF DOOR LENGTH
- ORBITER DESTRUCT SYSTEM (PREVENT HIGH SPEED IMPACT)
- BLAST CHANNEL/FIREWALL TO CONTAIN PLUME FROM WASTE PAYLOAD EJECTION SYSTEM MOTOR (MAINTAIN ORBITER STRUCTURAL INTEGRITY AFTER EJECTION)
- CREW ESCAPE PROVISIONS
- STRUCTURE AND LANDING GEAR STRENGTHENING FOR MAXIMUM PAYLOAD WEIGHT LANDING.

Figure 5.4-4: Unique Requirements for Launch Vehicles

5.5 **WASTE PAYLOAD SYSTEMS.** The primary objective of the waste payload system study effort was to trade a full range of waste payload protection system options in the areas of containment, radiation shielding, reentry protection, and impact protection in a systematic manner to define the best waste payload protection system for the space disposal mission. Both removable and integral radiation shield containment systems were studied.

DERIVATION OF PROTECTION SYSTEM REQUIREMENTS. Protection system requirements were derived in three steps. The first was establishment of a classification system for accident events. Accident end conditions were then identified and the results used in reviewing the safety requirements defined for the reference system. Recommended changes included reduction in the waste payload radiation specification from 2 rem/hr at 1m to 1 rem/hr and the addition of orbiter crash conditions to the waste payload design conditions. These additions, plus the recommendations of the previous study, provided the basis for design of the waste payload protection system.

WASTE PAYLOAD PROTECTION TRADE STUDIES. Protection system trade studies were used to determine the best approach to meeting the protection system requirements. The effort began with definition of protection system options. Analyses were conducted in the areas of radiation shielding, ejection from the launch vehicle, shield removal, and waste payload thermal characteristics.

Protection Options. Options for protection of the waste payload are illustrated in Figure 5.5-1. Options exist in the choice of radiation shielding material, with the prime choice being between the use of uranium or steel for radiation protection. A second option is removal of the radiation shield at some point in the mission to reduce the amount of mass carried all the way to the destination. A final option is providing the capability to eject the waste payload in the event of a launch vehicle malfunction. Studies were conducted in each of these areas to determine which of these options would be carried to the total system trades in Task 8.

Containment/Radiation Shielding Trades. The primary objective of the shielding and containment study was to determine whether shielding and containment could

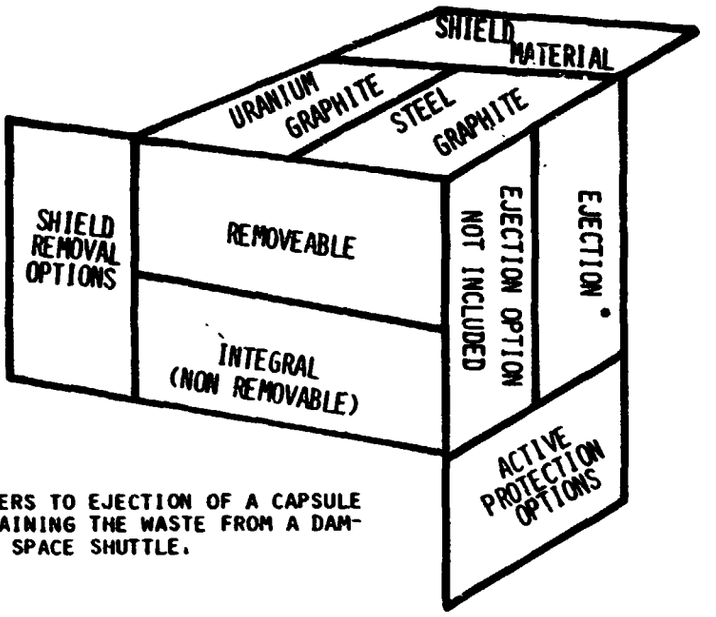


Figure 5.5-1: Waste Payload Protection Options

be integrated in a synergistic manner to maximize protection of the waste payload while minimizing total system mass. A series of trades were conducted to evaluate the characteristics of candidate isotropic and composite shielding/containment configurations. (Composite shields use separate materials for gamma and neutron attenuation.) Results of the analysis of isotropic and composite shields based on the reference 5,075 kg cermet waste form indicate:

1. Composite shields are lighter than isotropic shields due to their increased efficiency in reducing the neutron dose rate.

2. Water is the most effective moderator considered.

3. The lightest configuration by a narrow margin was the uranium/water composite shield at 11,520 kg. The next lightest was the uranium/graphite composite shield at 12,251 kg. The heaviest shield at 16,806 kg was the steel/graphite composite shield. This option was judged to be the safest.

The final shield evaluation considered survivability, mass, and availibility of shield material. Survivability considerations ruled out water as a moderator. Water could be lost too easily. A comparison of the remaining candidates showed that launch vehicle manifesting constraints and maximizing waste payload surviv-

ability both favor the composite steel and graphite shield. Accordingly, the steel/graphite composite shield was adapted as the sole candidate for further consideration in the study.

Ejection From Launch Vehicle. Ejection option effects on launch accident conditions are immediately apparent. Staying with the orbiter requires surviving the most rigorous possible environments. Ejection simplifies the survival problem but cannot be made 100% reliable. Ejection system reliability was examined using a review of data on USAF's experience with escape capsules used for aircraft crew escape. The study illustrates that ejection systems on the whole are very reliable, but problems cannot be totally eliminated.

Even though not 100% effective, the risk reduction effect of an ejection system can still be substantial. An ejection system which is not 100% reliable cannot be relied on to decrease the magnitude of the maximum insults delivered to the waste payload, but can be a valid means of reducing the frequency of occurrence of insults and thus the probability of a catastrophic accident event. The need for such a reduction cannot be established until the reliability of launch systems is more firmly established. In conclusion, the lack of 100% ejection system reliability implies that the waste payload system must survive all accident environments without ejection to ensure against catastrophic accident events, but ejection can still contribute to overall risk reduction. This conclusion led to the recommended addition of an Orbiter crash specification to the waste payload design accident requirements.

Shield Removal Trades. Primary issues in shield removal include structural integrity, thermal effects, EVA access, mechanization, and weight. The integral or nonremovable shield is superior in every area except weight. The need to evaluate the balance between risk and performance required consideration at the total system level in Task 8. The removable shield was evaluated only in combination with the ejectable waste payload protection system. The increased survivability of integral shield concepts allowed their evaluation both with and without the ejectable waste payload protection system.

Protection System Option Evaluation Conclusions. As a result of the protection system option trade studies, the following ground rules were adopted.

1. Steel/graphite composite radiation shields were adopted for all options.

2. Both integral and removable shields were evaluated for total system impact in Task 8.

3. Integral shield options were evaluated both with and without ejectable waste payload protection systems; removable shield options were evaluated only in conjunction with use of an ejectable waste payload protection system.

WASTE PAYLOAD CONFIGURATION DESCRIPTION. Using the results of the waste payload system trades in the areas of radiation shielding, shield removal and thermal analysis, integral and removable shield configurations were characterized by drawings and parametric weight estimating relationships. A dual waste payload system was also defined for use in dual launch concepts.

Integral Shield Configuration. The integrally shielded waste payload general arrangement is illustrated in Figure 5.5-2 for the reference 5075 kg cermet waste form. A primary feature is the high strength steel gamma radiation shield and primary container. The shield is welded into a one piece integral shell around the cermet waste form. The total individually shielded waste payload system is approximately 1500 mm in diameter and has a mass of 16,806 kg.

Removable Shield Configuration. The removable shield configuration is also illustrated in Figure 5.5-2. The composite radiation shield is similar in thicknesses to be integral shield. The cermet waste form is enclosed in a separate 64 mm thick reinforced stainless steel container; trunnions are set in the cermet and protrude through the radiation shield for independent support of the waste form during shield removal. Shield removal is mechanized by offset hinges which allow the two articulated segments to swing open, providing sufficient clearance for waste form removal. The shield is overlapped at the joints to contain radiation leakage. Overall size and shape of the removable shield are similar to the integral shield. The diameter of the removable waste form is approximately 1140 mm.

Dual Waste Payload. The dual waste payload configuration is illustrated along

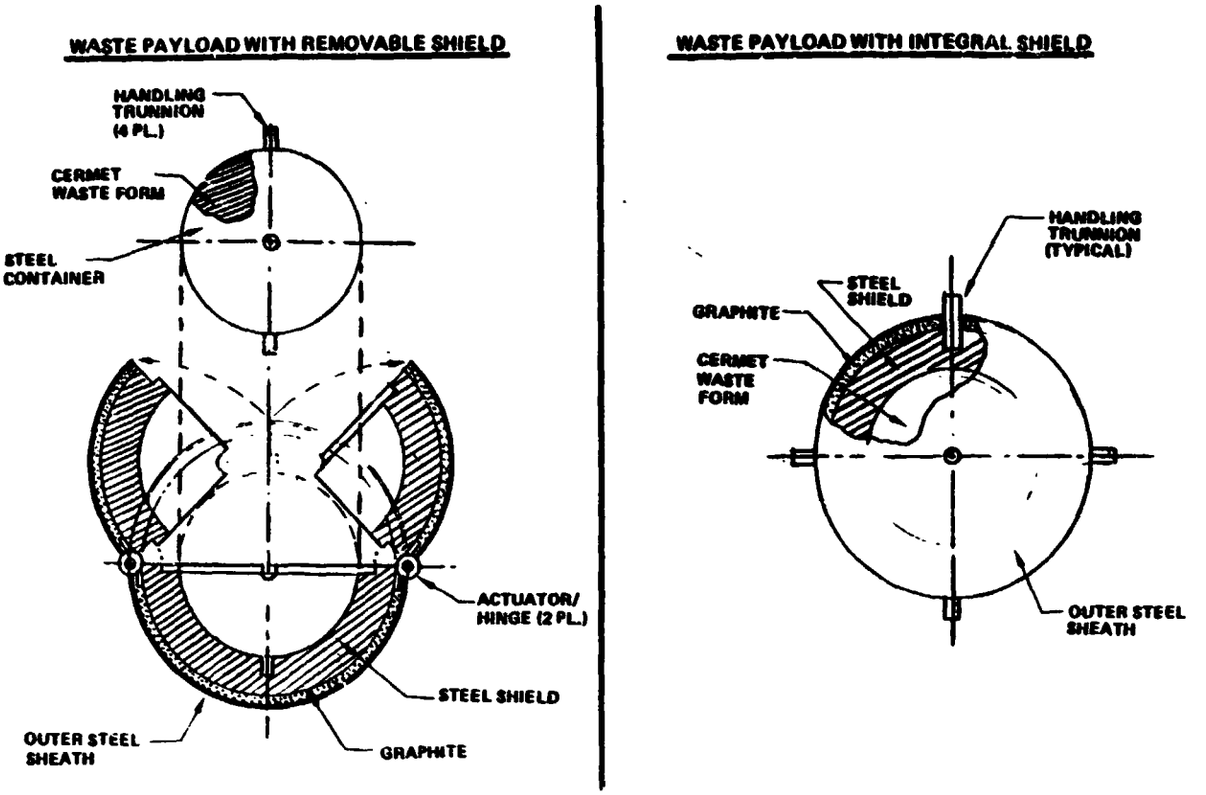


Figure 5.5-2: Candidate Waste Payload Configurations

with key dimensions in Figure 5.5-3. Two of the integrally shielded waste payloads are connected by a titanium inter-payload support structure. Total mass

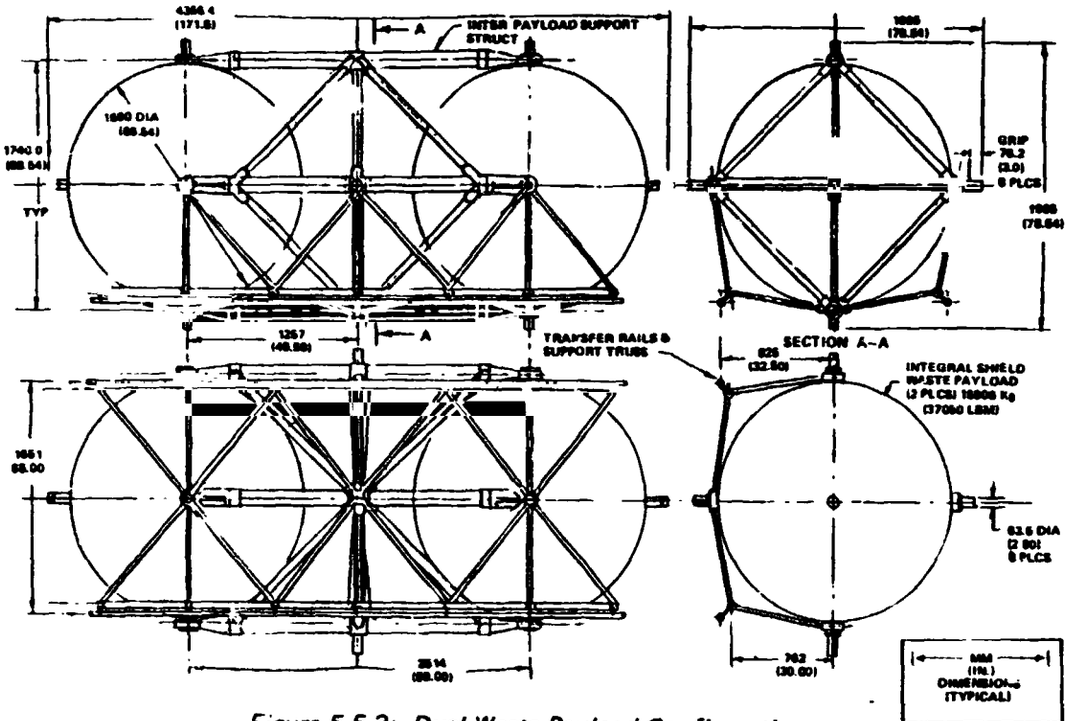


Figure 5.5-3: Dual Waste Payload Configuration

of the inter-payload support structure, including guide rails and associated hardware, is calculated at 136 kg. Support trunnions on the inter-payload support structure and outboard trunnions on the two waste payloads provide the structural interface with the flight support system. Guide rails are incorporated to aid payload transfer operations.

Waste Payload Weight Estimating Relationships. Weight estimating relationships for the waste payloads including waste form containment and shielding were prepared by optimizing shields for three waste form masses and using the resulting data points to prepare a curve showing the relationship between waste form mass and total waste payload mass.

These point designs were the basis for the weight estimating relationship curves plotted in Figure 5.5-4. The ordinate shows the mass of the composite waste payload including containment and radiation shield. The mass of the waste form is plotted on the abscissa. Individual curves are shown for the removable shield and the integral shield. Waste payload mass is slightly higher for the removable shield due to mass included to account for hinges and structural closure.

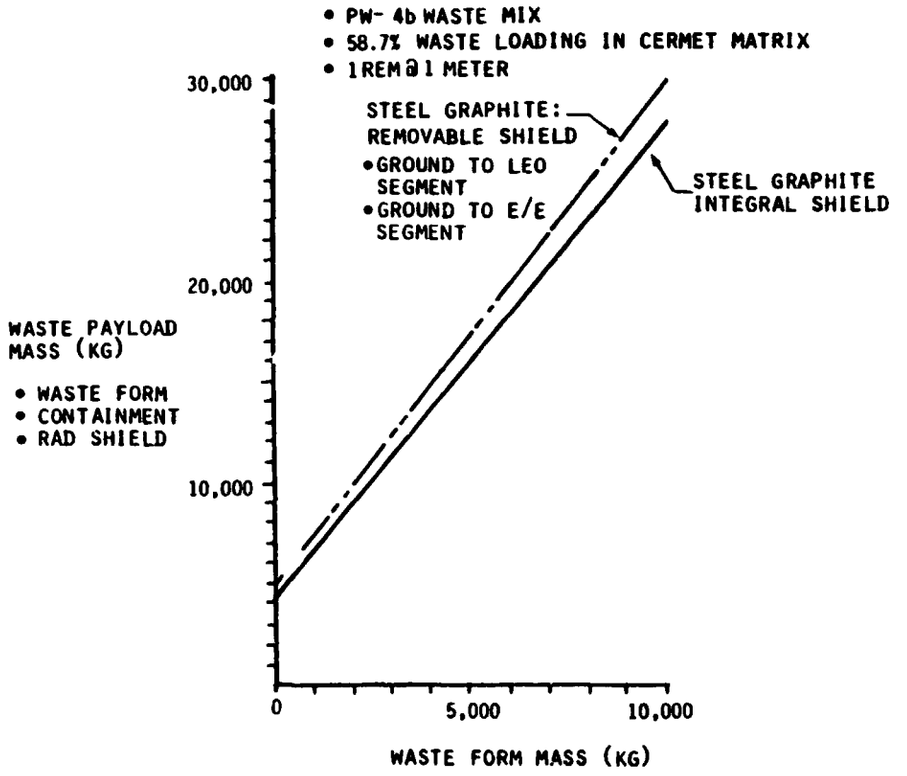


Figure 5.5-4: Waste Payload Weight Estimating Relationships for Selected Shield Options

WASTE PAYLOAD PROTECTION SYSTEM DEFINITION. The objective of the waste payload protection system is. 1) to provide intact recovery of the waste payload, restricting the consequences of any space transportation system accident to the non-critical accident category, 2) to minimize the expense of waste payload reflight by preventing damage to the waste payload due to transportation system accidents; and 3) to decrease the probability of a catastrophic accident.

The basic waste payload protection system has been well characterized in previous studies by the Marshall Space Flight Center, and the primary effort in this study was limited to characterization of the system for use in Task 8 total system trades. Accordingly, and in accordance with the ground rules specifying maximum use of past studies, the MSFC generated data for the waste payload protective system was used to provide parametric weight estimated relationships for the waste payload protective system.

Results are shown in Figure 5.5-5 which plots the total mass of the waste payload protective system against the mass of the waste payload carried from ground to low Earth orbit. Two curves are shown, the bottom one being the mass

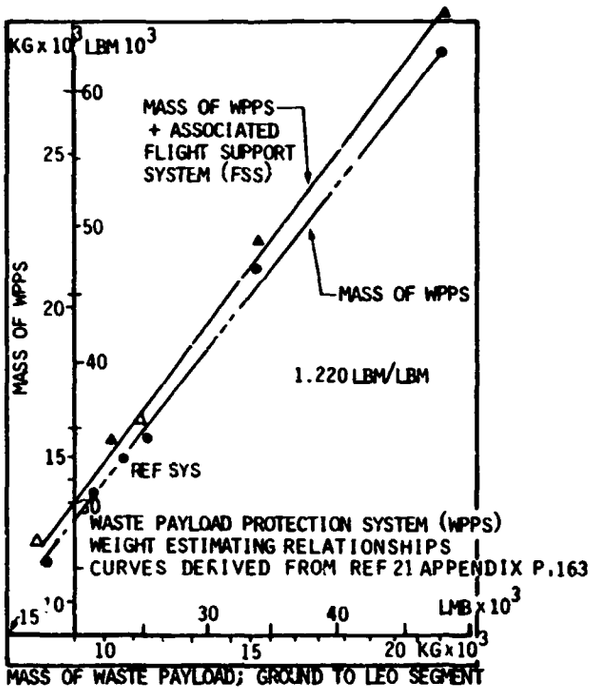
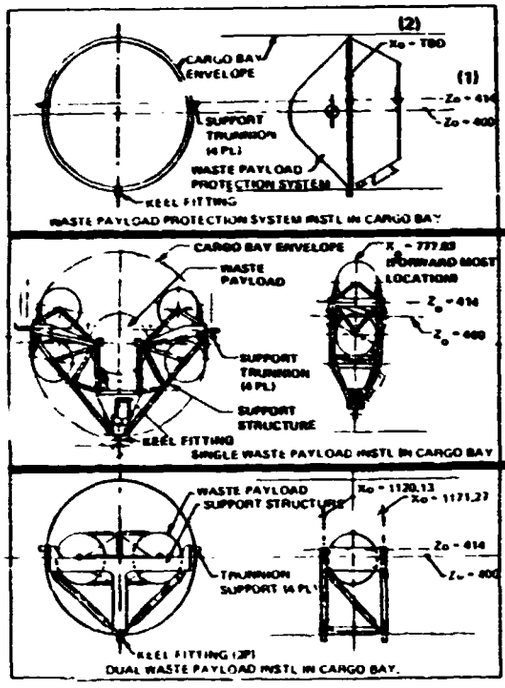


Figure 5.5-5: Waste Payload Protection System Parametric Mass Characterization

of the waste payload protective system by itself and the top curve being the mass of the waste payload protective system plus its associated flight support system. These curves were used for characterization of waste payload protective system mass in the total system studies in Task 8.

5.6 FLIGHT SUPPORT SYSTEMS. Primary objectives of the flight support system task were to identify system requirements for support of waste payloads in the launch vehicle cargo bay; to define and evaluate flight support systems concepts; and to characterize the resulting flight support systems. Because of the earlier work on flight support systems applicable to waste payloads protected by a waste payload protection system during launch, the effort in this study was concentrated on defining flight support systems for waste payload systems not requiring an ejectable waste payload protection system.

The task was initiated with a review of flight support system requirements from past studies, aimed at identifying the peculiar requirements imposed by alternate waste payload concepts and launch vehicles. This review lead to identification of concepts for and characterization of flight support systems for 1) the hardened waste payload not using ejection and 2) the dual payload cargo manifest required by dual launch options. The resulting concepts were characterized by drawings and preliminary mass statements for use in the total system evaluations conducted in Task 8. Figure 5.6-1 illustrates the concepts developed; weights and key dimensions are tabulated for each concept.



CONFIGURATION	USED ON OPTIONS	MASS (KG)	LENGTH (M)
WASTE PAYLOAD SUPPORTED BY WASTE PAYLOAD PROTECTION SYSTEM (WPPS)	DL-1 (3) SL-1 SL-3 SL-4 SL-5 SL-7 SL-8 SL-9	CARRIED AS PART OF WPPS	3.51 (ALL OPTIONS)
SINGLE WASTE PAYLOAD IN CARGO BAY	SL-2 SL-6 SL-10	1927* 2707* 4386*	1.40 1.40 1.80
DUAL WASTE PAYLOAD IN CARGO BAY	DL-2	5017*	2.7M (APPROX)

(1) ORBITER WATERLINE (REF.)
(2) ORBITER STATION (REF.) *INCLUDES COOLANT
(3) FOR OPTION DEFINITION SEE SECTION 5.7

Figure 5.6-1: Waste Payload System Options Requiring Flight Support

5.7 **SYSTEMS INTEGRATION AND EVALUATION.** Primary objectives of the systems integration and evaluation task were to integrate the systems resulting from Tasks 2, 3, 5 and 6 into alternate total system concepts, to define these candidate concepts in terms of performance and risk, and to trade the total systems in the areas of risk and performance to select alternative systems of high merit for further definition.

SYSTEM INTEGRATION. The systems integration effort was accomplished in two steps. In the first phase, studies were surveyed to ensure identification of all space system concepts. In the second phase, the "winners" of trades on launch vehicles and orbit transfer systems were arrayed against shield removal options and waste payload protection options identified in Task 2 to define a matrix showing reasonable alternative space system options for further analysis. The resulting matrix is illustrated in Figure 5.7-1. Options are desig-

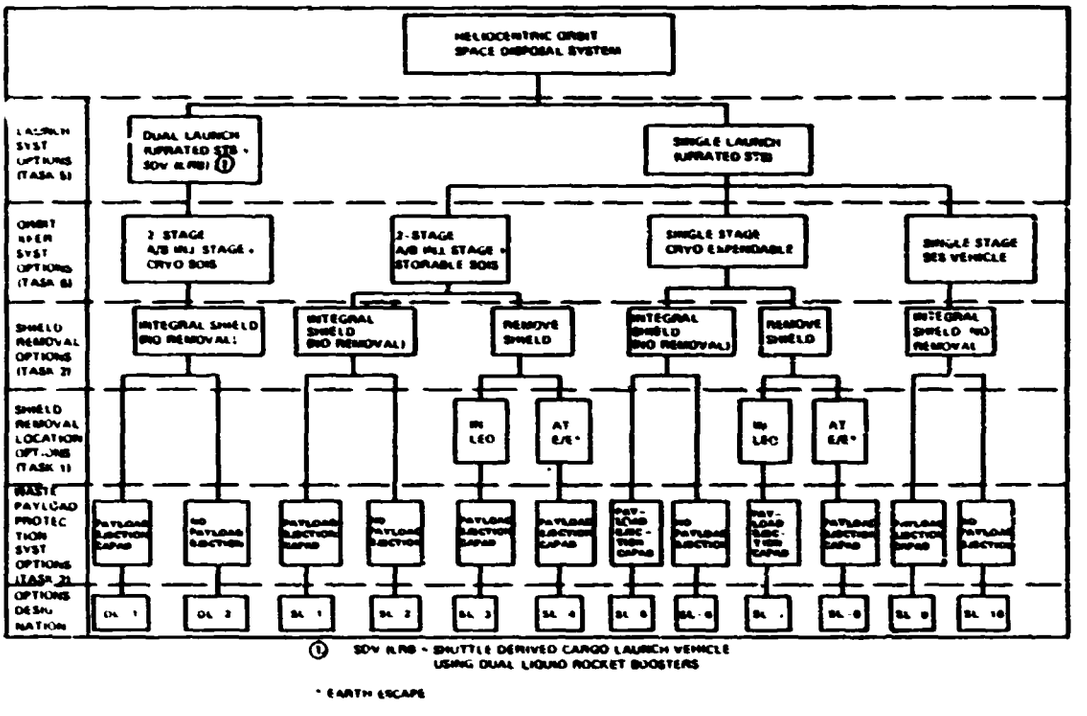


Figure 5.7-1: Selection of Options for Evaluation

nated by number on the bottom row. The description of each option can be obtained by following one branch of the trade tree reading from the top down. (As an example, Option DL-1 is a dual launch option using a two stage orbit transfer system composed of an aerobraked reusable injection stage and a cryogenic SOIS. This option has an integral shield waste payload system and makes use of an ejectable waste payload protection system.)

The space system options shown represent the complete range of reasonable alternative systems for space disposal, resulting from combining the highest performance orbit transfer options, the most cost effective launch vehicle options, and all viable waste payload protection options.

SYSTEM PERFORMANCE EVALUATION. The alternative space systems defined were evaluated for performance in terms of payload delivered per mission and cost. Performance was determined from the vehicle performance parametrics and BAC performance and mass estimating codes. Cost was calculated on the basis of average cost per flight based on launch vehicle production and operations cost from past studies, orbit transfer system operations based on estimates obtained in the phase A OTV study conducted by Boeing for MSFC. Production cost for orbit transfer systems was based on phase A derived unit cost for orbit transfer system hardware. The output of the system performance evaluation was a figure of merit expressed in recurring dollars per kilogram of waste form delivered to destination.

Figure 5.7-2 presents a comparison of the figures of merit for the 12 options considered. Costs are normalized to the cost of the reference system,

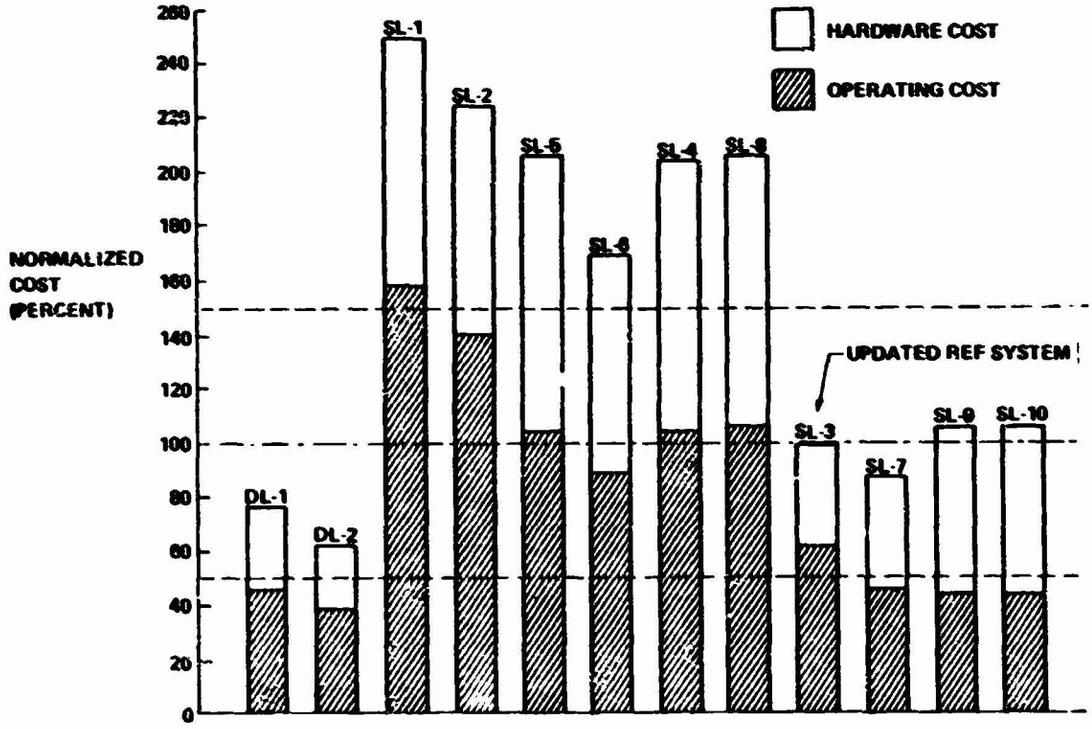


Figure 5.7-2: Comparison of Total System Performance

identified as system SL-3. The dotted lines above and below the cost shown for the reference system represent cost increases and decreases of 50%, respectively. To allow for the uncertainty in cost and performance estimates, only systems within the band were considered essentially equivalent in cost to each other and to the reference system. Systems falling outside the band were considered to be significanty more costly than the reference. It is apparent that only five options fall within this category, not counting the updated reference system. The two dual-launch systems (DL-1, DL-2) which take advantage of economies of scale and the higher efficiency of the cryogenic placement stages are the lowest cost options; the two electric vehicles (SL-9, SL-10) which utilize the high specific impulse of electric propulsion can, for all practical purposes, be considered equivalent in cost to the updated reference system. Both of these systems carry the waste payload shielding all the way to the destination. This represents a significant decrease in risk compared to the reference system. The single-stage cryogenic propellant option (SL-7), unlike the previously mentioned options, removes all waste payload protection in low Earth orbit. This option is equivalent in risk to the reference concept.

SYSTEM RISK EVALUATION. Risk evaluations were conducted to rank the alternate systems considered with respect to the risk of the updated reference system (option SL-3). Relative risk for the alternate systems was evaluated by comparing tne risk reduction provisions incorporated in each option.

Figure 5.7-3 presents the qualitative risk ranking for the 12 options considered. Four differentiable risk levels were identified based on the number of

			DEFINING CRITERIA (RISK AMELIORATION)			
	OPTION	RISK LEVEL	INTEGRAL SHIELD (HARD PAYLOAD)	WASTE PAYLOAD EJECTION CAPABILITY	"DRY" PAYLOAD BAY? (NO PROPELLANTS)	PROTECTED DURING CONTINGENCY RE-ENTRY
LOWEST RISK	DL-1, SL-9	1	YES	YES	YES	YES
	SL-1, SL-5	2	YES	YES	NO	YES
	DL-2, SL-10	2	YES	NO	YES	YES
	SL-2, SL-6	3	YES	NO	NO	YES
	SL-4, SL-8	3	NO	YES	NO	YES
HIGHEST RISK	SL-3, SL-7	4	NO	YES	NO	NO

Figure 5.7-3: Qualitative Risk Ranking

risk reduction provisions incorporated in the option. Risk level 1, shared by options DL-1 and SL-9, incorporates all provisions for risk reduction. Risk level 2, shared by options SL-1, SL-5, DL-2, and SL-10, indicates options incorporating three of the four risk reduction provisions. Options designated as risk level 3 incorporate two of the four risk reduction provisions; and options SL-3 and SL-7, at risk level 4, represent the highest risk systems considered. All options except SL-7 are lower in qualitative risk than the updated reference system (SL-3).

SPACE SYSTEM SCREENING FOR PERFORMANCE AND RISK. A final screening was conducted to combine the performance and risk evaluations to select alternative concepts possessing high merit. Five criteria were identified for the total system evaluation:

1. Technical feasibility
2. Risk
3. Cost
4. Environmental impact
5. Long term risk

Three of these criteria are not differentiators. Technical feasibility was established in Tasks 2, 5, and 6 for all of the options considered. Environmental impact is proportional to the number of launches and as such is measured by the system performance. Fewer launches equal less environmental impact. Long term risk was screened in Task 7; the destination selected, 0.85 AU heliocentric orbit, had the lowest long term risk of any destination identified.

The key remaining criteria which differentiate concepts are relative risk and performance. Risk criteria were evaluated by relative ranking. Systems having a risk equal to the reference system were rejected. Systems possessing a risk less than the reference system were judged acceptable. Performance was also evaluated relative to the reference system. Systems whose cost per mission was more than 150% of the reference system cost per mission were rejected. Systems possessing costs per mission judged less than or equal to that of the reference system were accepted.

Figure 5.7-4 presents the results of the screening for cost and risk conducted

CANDIDATE SYSTEM DESIGNATION	SUMMARY DESCRIPTION	WASTE FORM MASS DELIVERED PER FLT(KG)	RISK SCORE	COST SCORE	DISPOSITION AND COMMENTS
DL-1	DUAL LAUNCH A/B INJ + CRYO SOIS; WPPS + INTEGRAL SHIELD	8500	1	1	ACCEPT: TIES SL-9 FOR LOWEST RISK
DL-2	DUAL LAUNCH A/B INJ + CRYO SOIS; INTEGRAL SHIELD ONLY	10,150	1	1	ACCEPT: HIGHEST PERFORMANCE SYSTEM
SL-1	SINGLE LAUNCH A/B INJ + STORABLE SOIS; WPPS + INTEGRAL SHIELD	1599	1	X	REJECT: LOWEST PERFORMANCE SYSTEM
SL-2	SINGLE LAUNCH A/B INJ + STORABLE SOIS; INTEGRAL SHIELD ONLY.	1600	1	X	REJECT: PERFORMANCE < REF.
SL-3	SINGLE LAUNCH A/B INJ + STORABLE SOIS; WPPS + REMOVABLE SHIELD (LEO)	4082	X	1	UPDATED REFERENCE SYSTEM
SL-4	SINGLE LAUNCH A/B INJ + STORABLE SOIS; WPPS +, REMOVE SHIELD (E/E)	1977	1	X	REJECT: PERFORMANCE < REF.
SL-5	SINGLE LAUNCH, SINGLE STAGE CRYO LLOTV; WPPS + INTEGRAL SHIELD	2099	1	X	REJECT: PERFORMANCE < REF.
SL-6	SINGLE LAUNCH, SINGLE STAGE CRYO LLOTV; INTEGRAL SHIELD ONLY.	2500	1	X	REJECT: PERFORMANCE < REF.
SL-7	SINGLE LAUNCH, SINGLE STAGE CRYO LLOTV; WPPS + REMOVABLE SHIELD (LEO)	4989	X	1	REJECT: HIGHEST PERFORMANCE SINGLE LAUNCH SYSTEM BUT RISK = REF. SYSTEM
SL-8	SINGLE LAUNCH, SINGLE STAGE CRYO LLOTV; WPPS + REMOVABLE SHIELD (E/E)	2099	1	X	REJECT: PERFORMANCE < REF.
SL-9	SINGLE LAUNCH, SINGLE STAGE EXPENDABLE SES WPPS + INTEGRAL SHIELD	5449	1	1	ACCEPT: TIES DL-1 FOR LOWEST RISK
SL-10	SINGLE LAUNCH, SINGLE STAGE EXPENDABLE SES INTEGRAL SHIELD	5449	1	1	ACCEPT: PERFORMANCE + RISK ACCEPTABLE

Figure 5.7-4: Space System Screening

on the 12 candidate concepts. A 1 in the columns marked "risk score" and "cost score" constitutes acceptance. An X constitutes rejection. Based on these criteria, four systems were found acceptable. Option DL-1 is significantly less costly than the reference system and shares with option SL-9 the distinction of being the lowest risk concept identified. Option DL-2 is significantly lower in risk than the reference system and is the highest performance system considered. The other two systems accepted both use electric propulsion for the orbit transfer system. Option SL-9 is essentially equal in cost to the reference system but the risk is significantly lower. Option SL-10 also possesses identical costs to the reference with the same substantial risk benefits. These four systems were characterized as alternative systems possessing high merit.

5.8 LAUNCH SITE SYSTEMS. The primary objective of the launch site systems study was to define the impact of the selected space disposal options on facilities and operations at the Kennedy Space Center. In addition, an alternate site was to be selected and defined to allow an evaluation of the cost and benefits of accomplishing launches at a remote site. A review of the applicable references revealed that a great deal of work had been accomplished by the Marshall Space Flight Center in this area in previous studies. Accordingly, efforts in this study were confined to updating the Kennedy Space Center impact based on the Orbiter Fleet Size Study/KSC Launch Capability Analysis performed by KSC in 1980. Definition and evaluation of alternate sites were based on the Marshall Space Flight Center assessment.

Key findings are that the facilities impact of selected options is not sufficient to make it a primary differentiator between alternate space systems and that, while continued evaluation of an alternate site is warranted, this effort should be conducted as a part of the domestic and international affairs effort rather than as a part of the space system studies.

APPENDIX A

ACRONYMS

ACRONYMS

AU	astronomical unit
BAC	Boeing Aerospace Company
DDT&E	design, development, test, and evaluation
DOE	Department of Energy
EVA	extra vehicular activity
FSS	flight support system
HEO	high Earth orbit
HLLV	heavy-lift launch vehicle
IOC	initial operational capability
IUS	inertial upper stage
JSC	Johnson Space Center
KSC	Kennedy Space Center
LEO	low Earth orbit
LeRC	NASA Lewis Research Center
LLOTV	long-life OTV
LRB	liquid rocket boosters
MSFC	Marshall Space Flight Center
NASA	National Aeronautics and Space Administration
NPFF	nuclear payload preparation facility
ONWI	Office of Nuclear Waste Isolation
OTV	orbital transfer vehicle
RCS	reaction control system
RF	radiofrequency
RV	reentry vehicle
SDV	shuttle-derived vehicle
SEPS	solar electric propulsion system
SES	solar electric stage
SOIS	solar orbit insertion stage
SRB	solid rocket boosters
SSTO	single stage to orbit
STS	space transportation system
WPPS	waste payload protection system